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AFFDL-TR-76-72

AIRCRAFT STRUCTURAL DESIGN HANDBOOK FOR LOW-COST MAINTENANCE AND REPAIR

ROCKWELL INTERNATIONAL
LOS ANGELES AIRCRAFT DIVISION
LOS ANGELES, CALIFORNIA 90009

MARCH 1977

TECHNICAL REPORT AFFDL-TR-76-72
FINAL REPORT FOR PERIOD AUGUST 1974 - MAY 1976

AIR FORCE FLIGHT DYNAMICS LABORATORY
AIR FORCE WRIGHT AERONAUTICAL LABORATORIES
AIR FORCE SYSTEMS COMMAND
WRIGHT-PATTERSON AIR FORCE BASE, OHIO 45433

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This technical report has been reviewed and is approved for publication.

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FOR THE COMMANDER

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20. ABSTRACT (Continue on reverse side if necessary and identify by block number)

→ This document has been prepared to serve a growing need in the military to reduce aircraft structural maintenace costs to a more reasonable level commensurate with acceptable life-cycle costs. It is designed as an informative guide which will aid the aircraft designer in foreseeing maintenance problems and make proper trade-off evaluations to optimize the structural design for total life-cycle costs. → next page

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20. ABSTRACT (Continued)

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The handbook points up several examples of high maintenance cost items on existing in-service aircraft and suggests changes to substantially reduce the life-cycle cost. In addition, many other costly maintenance items discovered during visits to military and industry maintenance and repair facilities are cited which could have been avoided or substantially reduced by more cost-effective considerations for serviceability during design.

In this respect, the handbook includes, not only information on past problem areas in the form of "lessons learned," but recommended considerations during initial design of every aspect of structural development. Since corrosion damage repair was found to be one of the most costly maintenance items, a part of the handbook provides design information usable in its prevention. Also, since the handbook is directed primarily toward the development of military aircraft, a section is devoted to battle damage and design considerations to increase survivability and permit repairs to minimize downtime on the aircraft. Report No. AFFDL-TR-76-63" Low Cost Aircraft Structural Repair and Maintenance Study," documents the research and analysis conducted in the development of this Design Handbook.

FOREWORD

The Structures Division of the Air Force Flight Dynamics Laboratory (AFFDL) has sponsored a Low Cost Aircraft Structure Repair and Maintenance program under Air Force contract F33615-74-C-3101, Project No. 1368, Task No. 136802.

The work was performed by the Los Angeles Aircraft Division (LAAD) of Rockwell International and managed by Mr. Walter Dotseth. It was monitored by Mr. Clark Beck, AFFDL/FBS.

This publication is a product of the fifth and final phase of the program and includes data obtained from an industrywide literature search, from Air Force and Navy maintenance cost reporting systems, and from first-hand contacts and inspection surveys at five Air Force Air Logistic Centers (ALC), one Military Aircraft Storage and Disposition Center, three Naval Aircraft Rework Facilities, four commerical airlines maintenace facilities, and six major manufacturers of military aircraft. These inspection tours were made by Mr. W. E. Routh and Mr. R. W. Nickel of Rockwell accompanied by Mr. C. Beck of AFFDL/FBS in 1974 and 1975.

The objectives of this program were twofold:

1. To survey maintenance facilities and current repair reporting systems to establish high maintenance cost drivers and show, through actual design of repair modifications, specific examples of significant cost reduction potential.

2. To develop a design guide for structural designers to provide the necessary data for consideration during initial design which will result in lower cost maintenance and repair.

The information obtained in the literature search and maintenance facility visits plus a significant quantity of data generated by Rockwell engineers on this program, and for other programs in years past, has been culminated in the Aircraft Structural Design Handbook for Lower Cost Maintenance and Repair. The results have been detailed in sections 4 through 7 of the volume. Section 4, "Lessons Learned," provides numerous examples of high-cost maintenace items currently existing on service aircraft. Section 5, "Repair and Modification Design Concepts," provides suggested designs for modification of some of the higher cost examples in section 4 plus other suggested typical repair designs. Sections 6 and 7 provide general and detail design considerations to follow in initial design to prevent high maintenance costs.

Acknowledgment and appreciation is extended to those Air Force and Naval organizations, commerical airline companies, and aircraft manufacturers who provided the supplemental assistance necessary for this effort. The valuable information provided contributed directly to the objectives and results of this program.

Report No. AFFDL-TR-76-63 "Low Cost Aircraft Structural Repair and Maintenance Study," documents the Research and analysis conducted in the development of this design handbook.

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1.0 INTRODUCTION

In recent years, the operational cost of ownership of military aircraft has reached almost twice their initial acquisition costs. This is due largely to higher maintenance and repair costs of the system. By analysis of data from the Air Force IROS and AFM66-1 cost reporting systems over a recent peacetime period for 20 aircraft of all types, it was determined that structural maintenance costs constituted 25 to 30 percent of the costs for a total aircraft system. This verified the necessity for a program to identify problems and provide a design guide to reduce such trends on future aircraft and to provide guidance on means to reduce structural maintenance costs on existing aircraft.

This handbook has been prepared to provide information to the designer which can be used advantageously to improve new aircraft structure designs as well as future modification designs for in-service aircraft. The sources of data presented are from literature search of existing publications and from first-hand inspection of repair and maintenance facilities. During the course of inspection, contacts were made with responsible personnel to obtain primary cost drivers for aircraft structure maintenance and repair. These were coordinated with a review and analysis of computer data from the Air Force IROS and AFM66-1 maintenance cost reporting systems. A representative number of the high cost repair items were selected as a sampling for detail design of modifications to verify potential to significantly reduce existing costs.

Results of the inspection survey and the modification design effort are included in section 4, "Lessons Learned," and section 5, "Repair and Modification Design Concepts." Sections 6 and 7 are devoted entirely to general and detail design considerations to follow during initial design of new aircraft structure or in design of modifications for in-service aircraft in order to assure low-cost maintenance and repair.

The essential objective of this publication is to provide information and guidance to aircraft structural designers on techniques to minimize cost by proper selection of design features and repair considerations.

The information in this handbook is limited to existing types of metal structures. This is due to the existence of other programs sponsored by the Air Force Flight Dynamics Laboratory for the research and development of repair and maintenance information on adhesively bonded and composite type aircraft structures. Guidance will be provided on cost-effective design practices in separate design handbooks.

2.0 USE OF HANDBOOK

2.1 INTRODUCTION

This design handbook has been developed for use by aircraft structural designers to provide information and guidance on the methods to minimize or reduce the costs of structure maintenance and/or repair. It is organized into specific sections to provide the reader with background information, lessons learned, general design considerations, detail design considerations, repair or modification design considerations, and reference to additional information.

2.2 RECOMMENDED USE

It is recommended that the user first read the background information contained in section 3.0 to become acquainted with the importance and magnitude of the maintenance and repair efforts, and their costs, associated with military aircraft structures. This is to provide the reader with an appreciation of the importance that the general and detail design elements of an aircrafts structure can have on the cost of ownership to the user, and the emphasis that is being placed on life-cycle costs of aircraft systems.

Section 4.0 contains a listing of "lessons learned" by the military and commercial airlines on some of the most frequent and chronic types of discrepencies being experienced on metallic structures. They are intended to provide the reader with a knowledge of the types of structural design feature that should be avoided in new systems.

In section 5.0, General Design Considerations, information and guidance is provided on the basic structure design features that should be considered during design concept formulation of a new aircraft. Many maintenance and repair problems may be avoided or reduced in the operational life of an aircraft system by early consideration of these factors during its development.

In section 6.0, Detail Design Considerations, information and guidance is provided on candidate methods and features that may be incorporated into the detail design of aircraft structures to reduce maintenance and repair costs. Also contained in this section, is information and guidance on methods to evaluate the impact of improved maintenance or repair design features on the life-cycle cost (LCC) of an aircraft system.

In section 7.0, Repair Modification Considerations, information is provided on the factors that should be considered in developing improved low-cost repair or modification concepts on existing aircraft structures. It also contains examples of some typical repair and design modification concepts for normal operation and battle-damaged aircraft structures. These examples are intended to give the reader an awareness of the different types of approaches that may be considered and to stimulate his imagination in development of repair and/or modifications for a specific problem.

In section 8.0, References, a listing of the data sources used for the development of this design handbook are listed. They provide the reader with identification on the location of additional data.

3.0 BACKGROUND

The contents of this document are based on information obtained through an extensive literature search, the knowledge and design capability of many experienced structural designers and repair personnel, first-hand visits to the USAF Air Logistic Command repair depots, several Naval Aircraft Repair Facilities, and commercial airline maintenance centers. During these visits, structural problem area information and repair data were obtained on the primary types of service aircraft. These included fighter, trainer, attack, bomber, tanker, cargo, and troop transport aircraft. The commercial airline type of aircraft included most of the modern jet transports.

The data obtained and used to develop this manual are considered to represent a complete cross-section of the structural maintenance and repair action on present military service aircraft. The growing high cost of structure repair action can be reduced substantially by the judicious consideration, during initial design phase, of the serviceability aspect of the aircraft. The major factors involved are selection of structural material, methods of fabrication, structural assembly breakdown, joint design, maintenance access, environmental protection, and consideration of maintenance practices. Logistics of parts availability also may play an important part in maintenance costs.

4.0 LESSONS LEARNED

4.1 INTRODUCTION

During visits to the various maintenance and repair facilities, it became evident that, in many cases, the same design deficiencies were appearing on current aircraft structures that have occurred on previous generations of aircraft. In this chapter, a number of these repetitive problems are identified, so that the designers of future structures may take advantage of the lessons learned. Solutions for some of these problems are shown in section 7.0.

4.2 DESIGN DEFICIENCIES

Problems encountered at the repair facilities were primarily design deficiencies. They are grouped in five main categories:

1. Lubrication deficiencies (paragraph 4.2.1)

2. Corrosion protection deficiencies (paragraph 4.2.2)

3. Material selection deficiencies (paragraph 4.2.3)

4. Detail design deficiencies (paragraph 4.2.4)

5. Fatigue design deficiencies (paragraph 4.2.5)

4.2.1 LUBRICATION DEFICIENCIES

1. Main landing gear trunnion and pin

   Problem: Galling and corrosion were found frequently on MLG lower trunnion bearing and pin surfaces. (See figure 1.)

   Cause: Drainage of dirty water thrown up by the MLG wheels, from the forward face of the aft bulkhead and support fitting, entered the trunnion bearing surfaces through openings on the upper surface of the trunnion fitting and settled in the small gap between the pin and trunnion on the lower side. The water carried grit which, during taxiing and takeoff/landing operations on rough runways, eroded the chrome surface of the pin and the unprotected surface of the trunnion. (Refer to section 7, paragraph 7.3.1 for solution.)

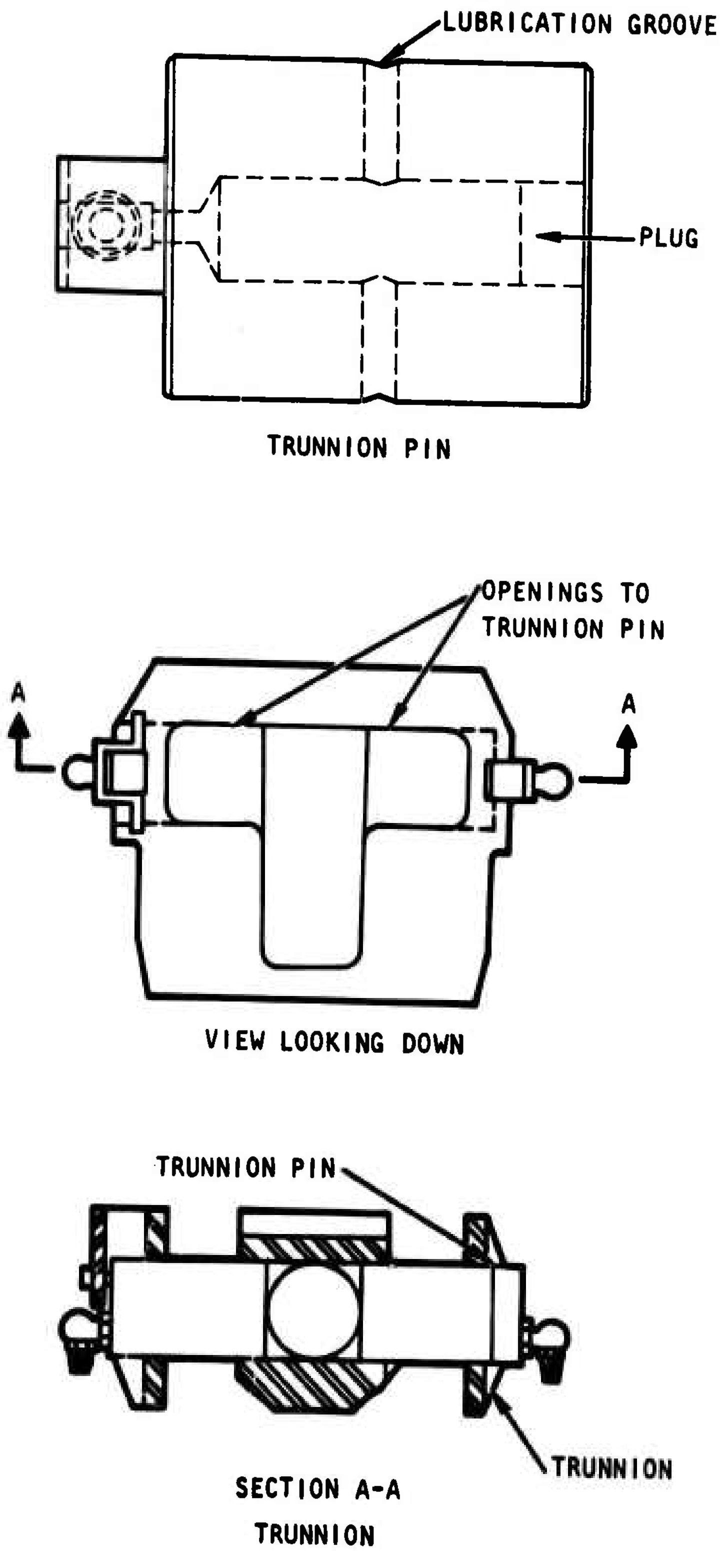


Figure 1. Main Landing Gear Trunnion and Pin

2. Moisture retaining bushings - MLG strut

Problem: Corrosion was found around Teflon bushings in the main landing gear strut.

Cause: The bonding material for the attachment of the Teflon bushing to metal strut is porous and retains moisture that permits corrosion to occur.

3. Lubrication provisions - horizontal stabilizer bearings

Problem: Horizontal stabilizer bearings were found to be corroded.

Cause: Only two grease fittings were provided to lubricate a large diameter bearing, making it very difficult to force out old moisture-contaminated grease all around the bearing.

4.2.2 CORROSION PROTECTION DEFICIENCIES

1. Inner to outer wing joint ribs

Problem: Severe surface corrosion has been encountered in the upper wing joint bath tub pockets for the attach bolts. The bolts and nuts have also become corroded. This condition is prevalent primarily on the upper bolting rib caps. (See figure 2.)

Cause: Leakage of water and anti-icing fluids around the cover fairing allows moisture to collect in the pockets of the bolting fittings. No provision exists for sealing, drainage, or drying of the area. (Refer to section 7, paragraph 7.3.2 for solution.)

2. Galley, lavatories, and urinal areas

Problem: Moisture migrates into the aircraft structure under the galley, lavatories, and urinal areas of various aircraft, causing corrosion of the seat and cargo tiedown tracks, floor beam caps, and supporting structure. Usually, these areas could not be inspected without removing permanent floor panels or external skins. (See figure 3 for a typical urinal installation in a bomber aircraft.)

Cause: Leakage or spillage of water or corrosive fluids is not prevented from seeping into areas where corrosion of structural members will occur. (Refer to section 7, paragraph 7.3.3 for design solution.)

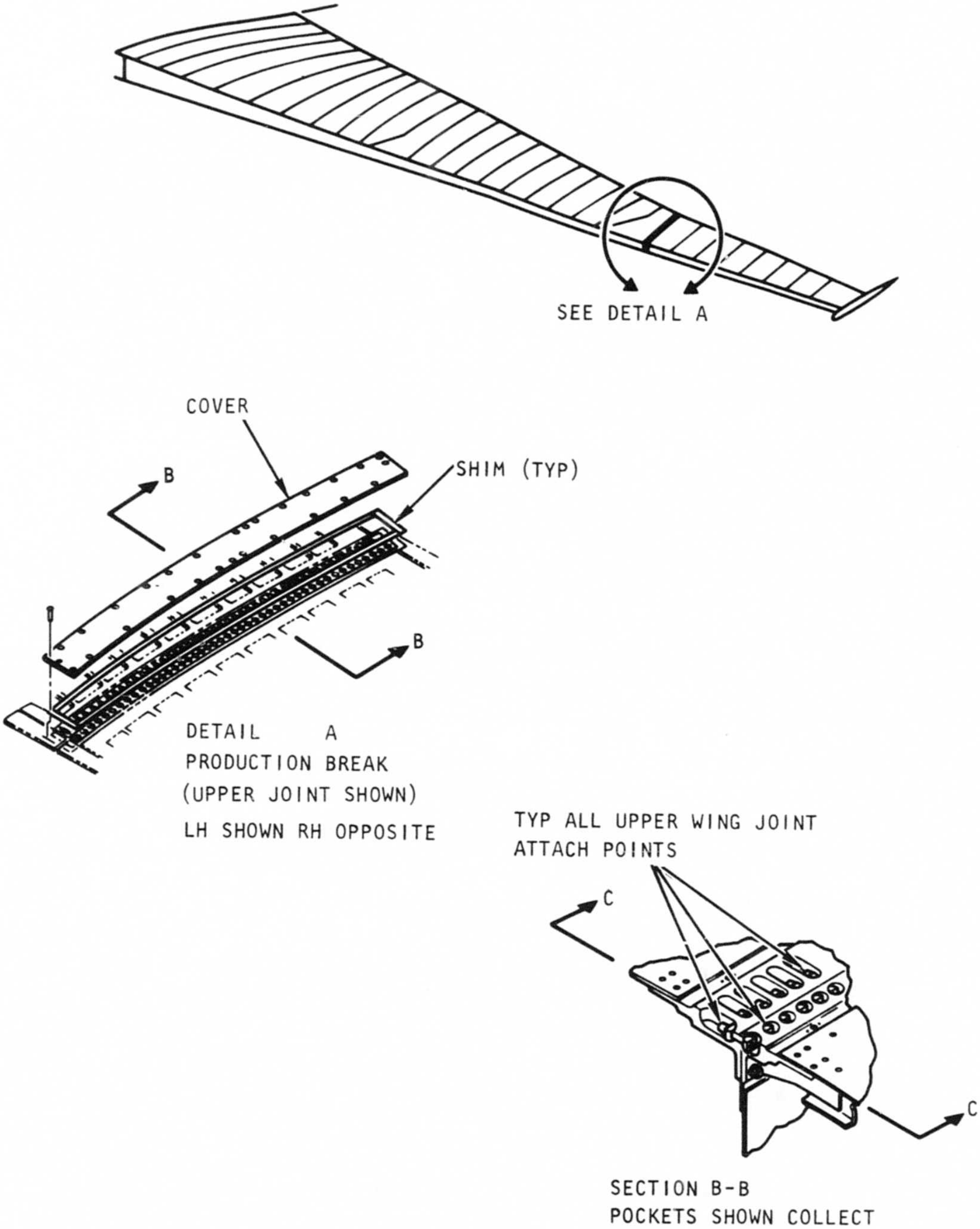


Figure 2. Inner to Outer Wing Joint Rib

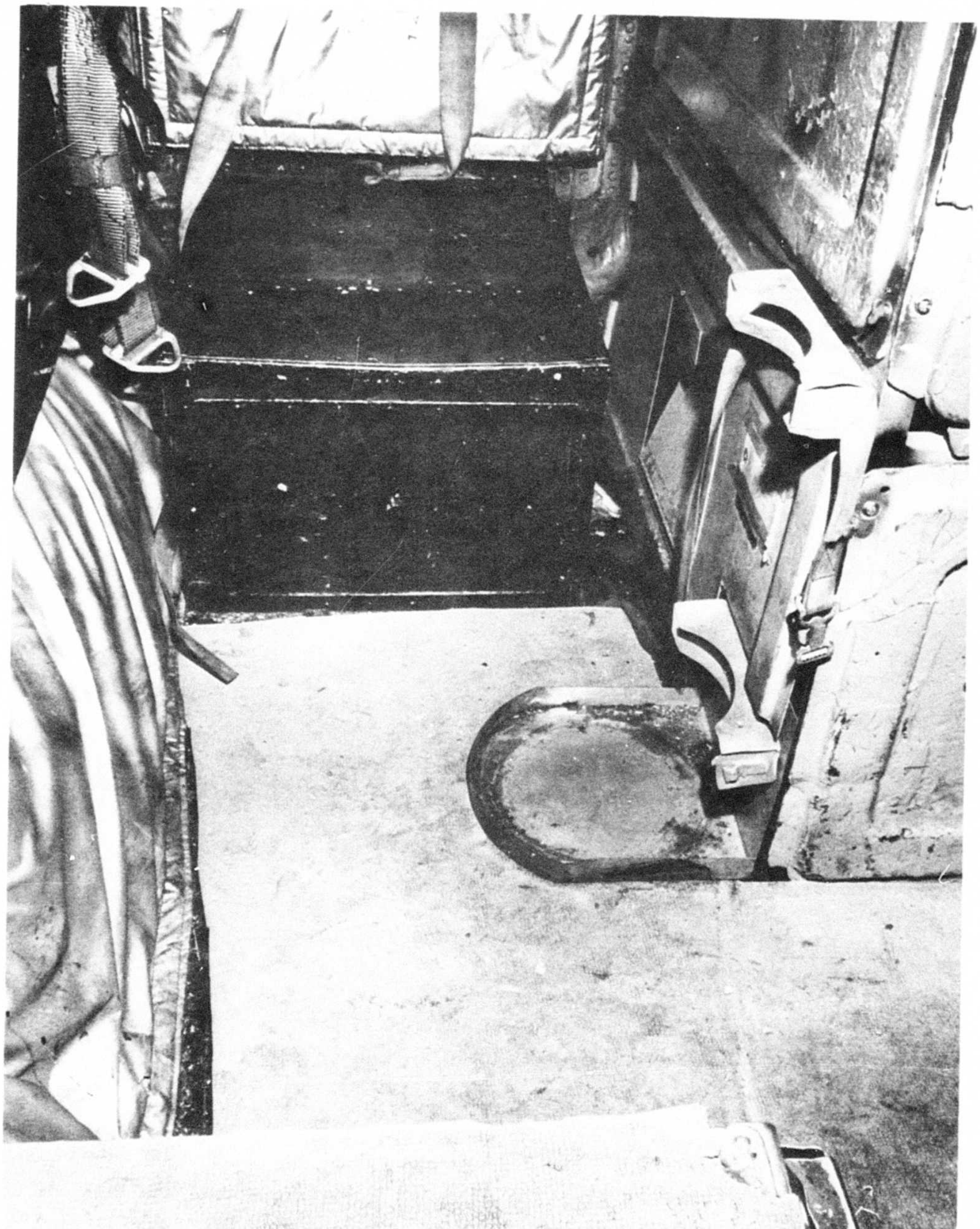


Figure 3. Typical Urinal Installation in a Bomber Aircraft
Urinal Removed for Clarity

3. Rain removal nozzle

Problem: Severe corrosion of a rain removal nozzle required frequent replacement. (See figure 4.)

Cause: There are two causes to this problem. The first is that the nozzle is constructed of magnesium material which is very susceptible to corrosion. The second is that the nozzle is mounted in the aircraft just forward of the windshield and flush with the outside mold line of the fuselage. This left the air outlet end of the nozzle exposed to rain, salt spray, fog, etc. Moisture entered the air outlet opening and ran down inside the nozzle where it collected and allowed corrosion to occur.

4. Cockpit canopy longeron

Problem: Chronic corrosion of cockpit longeron under a phenolic filler plate has been experienced. (See figure 5.)

Cause: The phenolic filler plate is bonded to an aluminum longeron. Moisture enters the fibers at the edge of the phenolic filler and propagates into the bond area where it is trapped in small voids and corrodes the metal.

5. Vertical stabilizer attach bulkhead

Problem: Cracking of a steel forged bulkhead for the vertical stabilizer front spar attachment was found to frequently occur.

Cause: Water becomes trapped in a pocket where the stabilizer front spar is attached. The trapped water freezes during flight and contributes to crack development.

6. Rudder upper hinge support

Problem: Frequent corrosion of rudder hinge upper support. (See figure 6.)

Cause: Entrapment of moisture, due to omission of drainage provisions, and 7075-T6 aluminum material combine to cause corrosion.

7. Main landing gear strut door

Problem: A fighter aircraft main landing gear strut door was found to experience frequent corrosion and cracking of the structural elements.

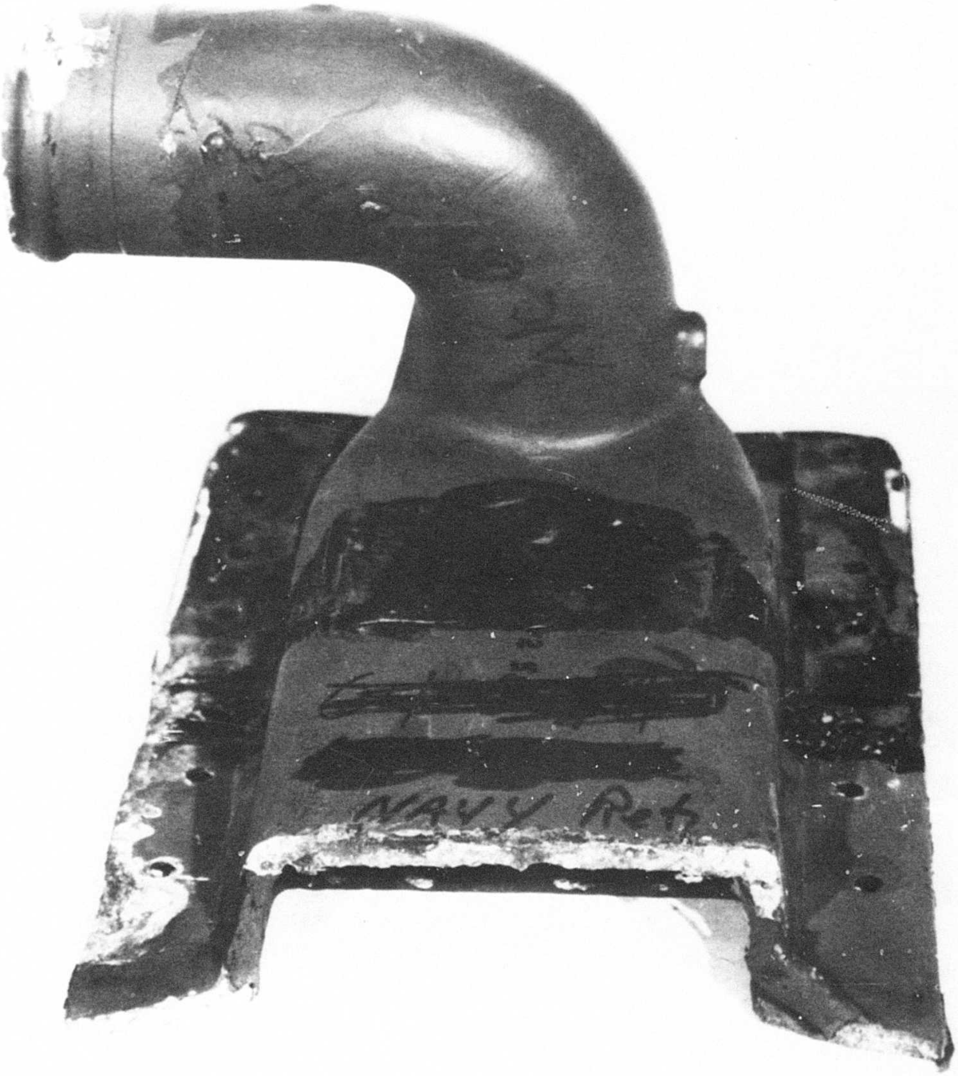


Figure 4. Corrosion of Rain Removal Nozzle

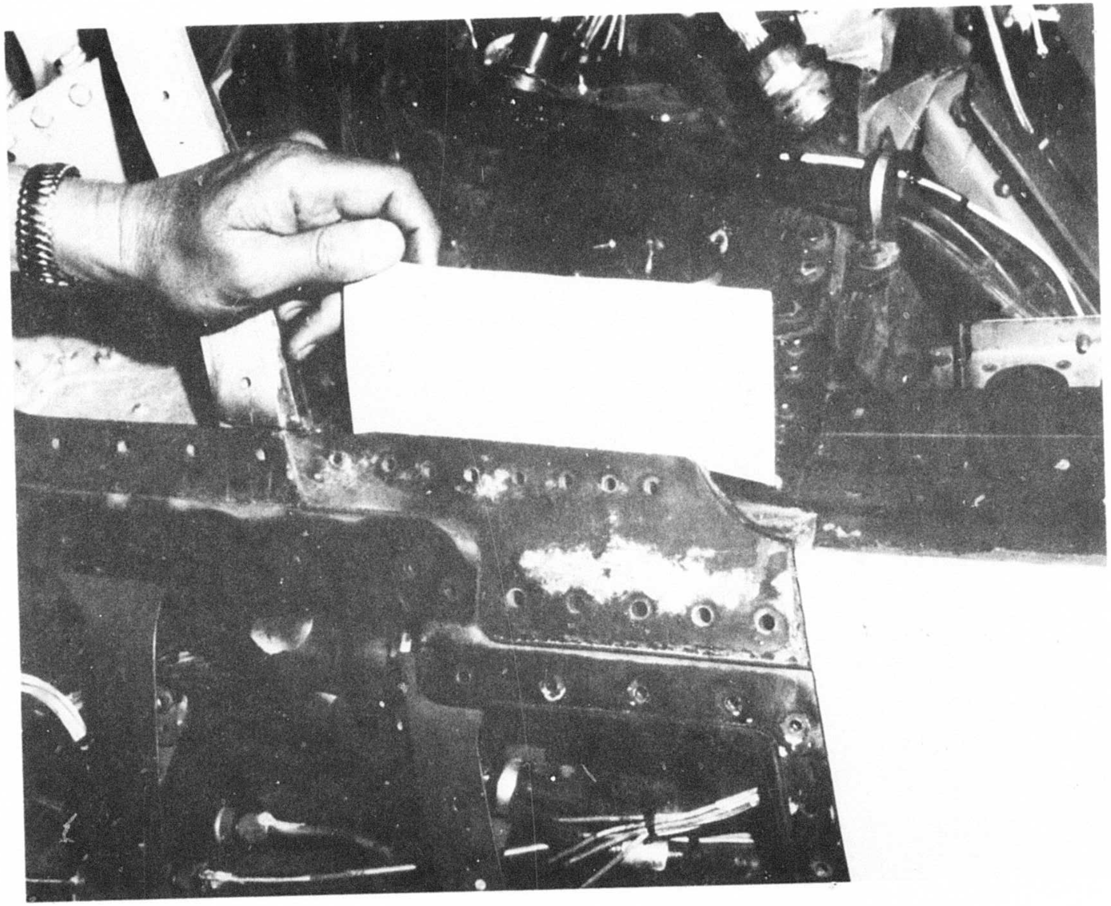


Figure 5. Corrosion of Cockpit Longeron

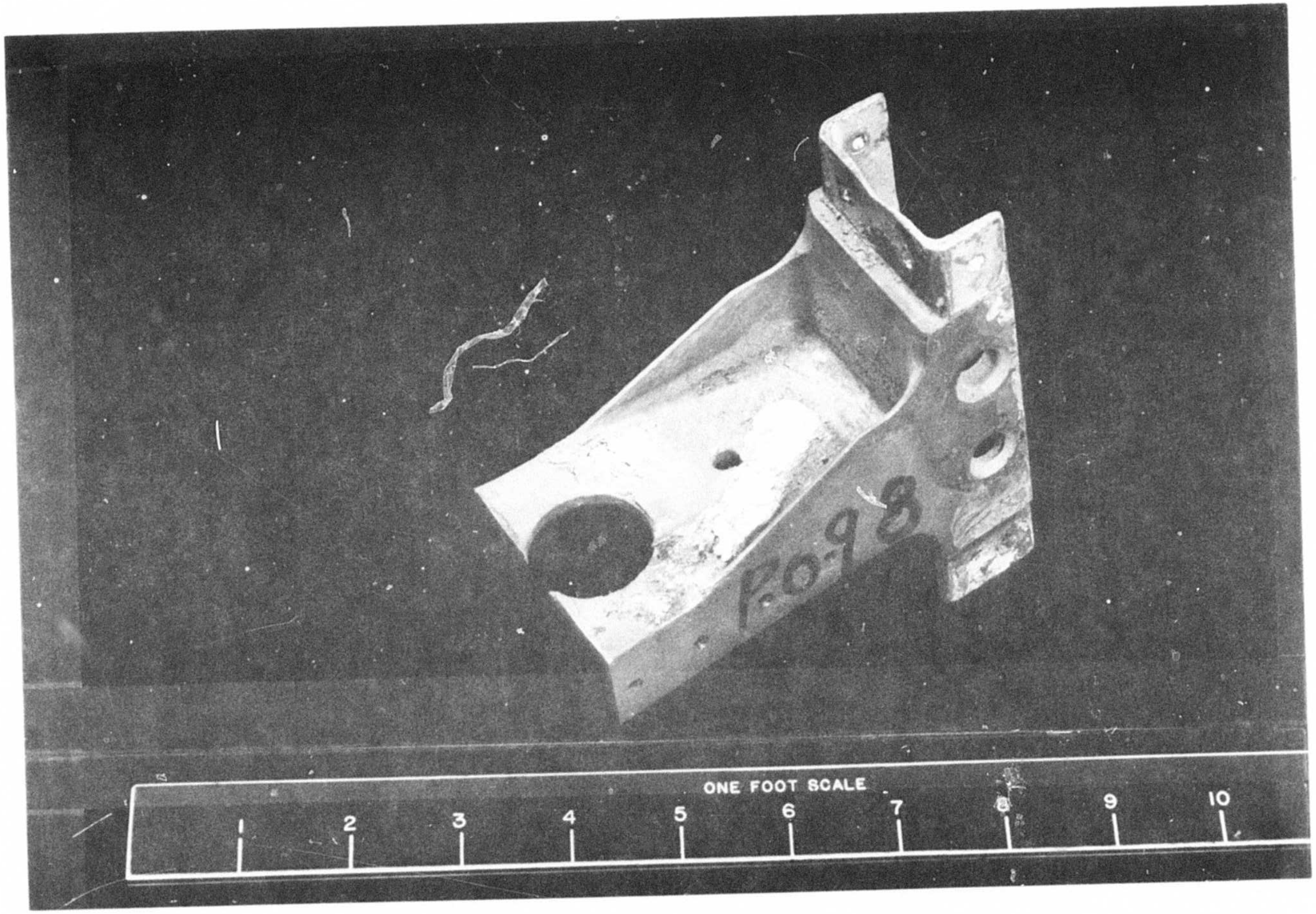


Figure 6. Corrosion of Rudder Hinge Support

Cause: Corrosion of the strut door main framework casting and inner and outer skins was caused by moisture being soaked up and trapped by the foamed-in-place filler material (REN).

8. Fail-safe strap - fuselage lower centerline

Problem: Corrosion of fail-safe strap and lower fuselage skin at lower centerline. See figures 7 and 8.

Cause: Fail-safe aluminum strap is bonded to aluminum fuselage skin with a cold bond material. Bonding material cracks and allows moisture to enter the faying surface and become trapped, which causes corrosion to develop.

9. Stabilator lower skin at joint rib

Problem: Corrosion of a stabilator lower skin was found to occur at the rib attach point. See figure 9.

Cause: Aluminum skin corrodes due to inadequate moisture drainage provisions and skin joint sealing.

10. Rudder root rib

Problem: Root rib at base of a rudder fitting was frequently found corroded.

Cause: Type 1048 sealant was used which hardens and pulls away from edges. This allowed water to enter and induce corrosion of the root rib.

4.2.3 MATERIAL SELECTION DEFICIENCIES

1. Wingtip fold rib

Problem: Failures of wingtip fold lower aft locking lug are being experienced. (See figure 10.)

Cause: Fatigue and stress corrosion cracking due to high stress loads and moisture.

2. Wing spar inboard section

Problem: Cracks are being experienced in rear inboard wing spar section. (See figure 11.)

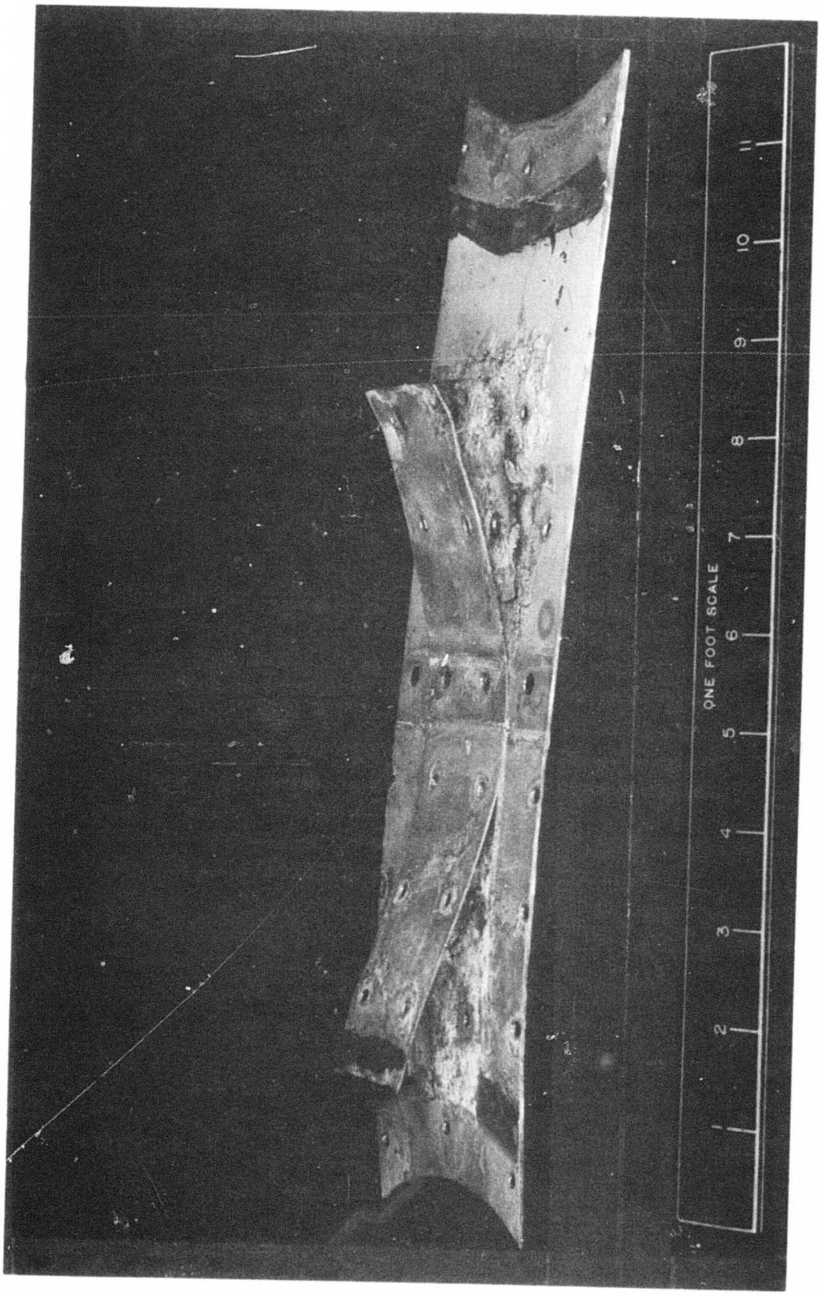


Figure 7. Corrosion of Bonded In-place Fail-Safe Strap

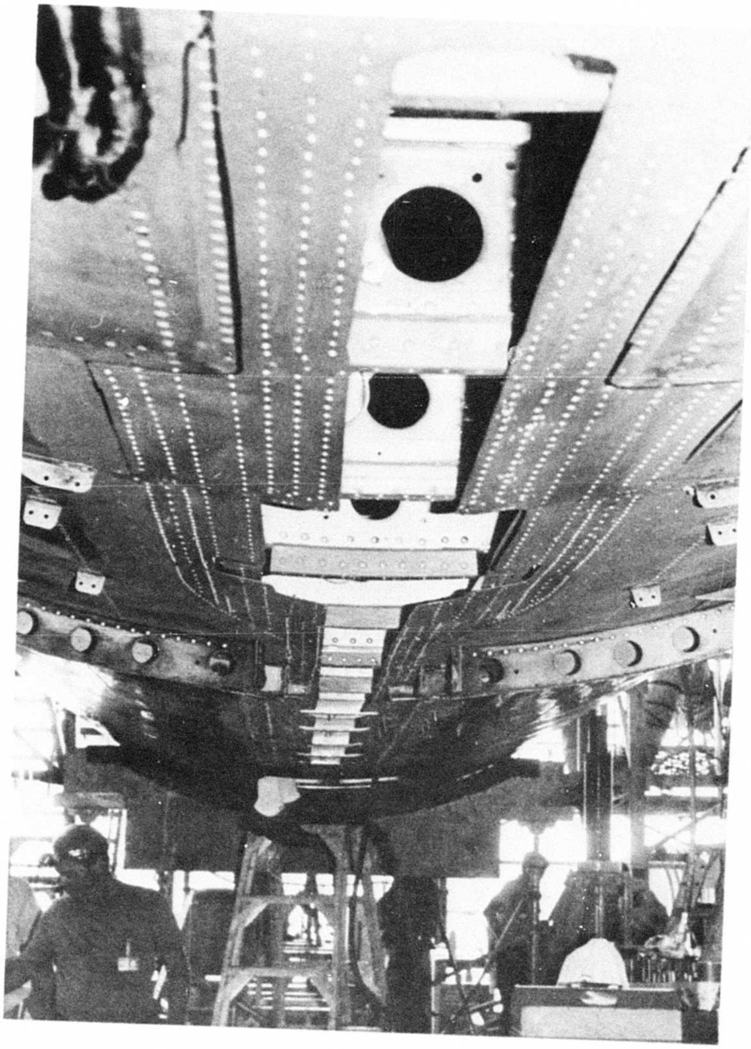


Figure 8. Extent of Repair Caused by Corosion of Bonded Fail-Safe Strap

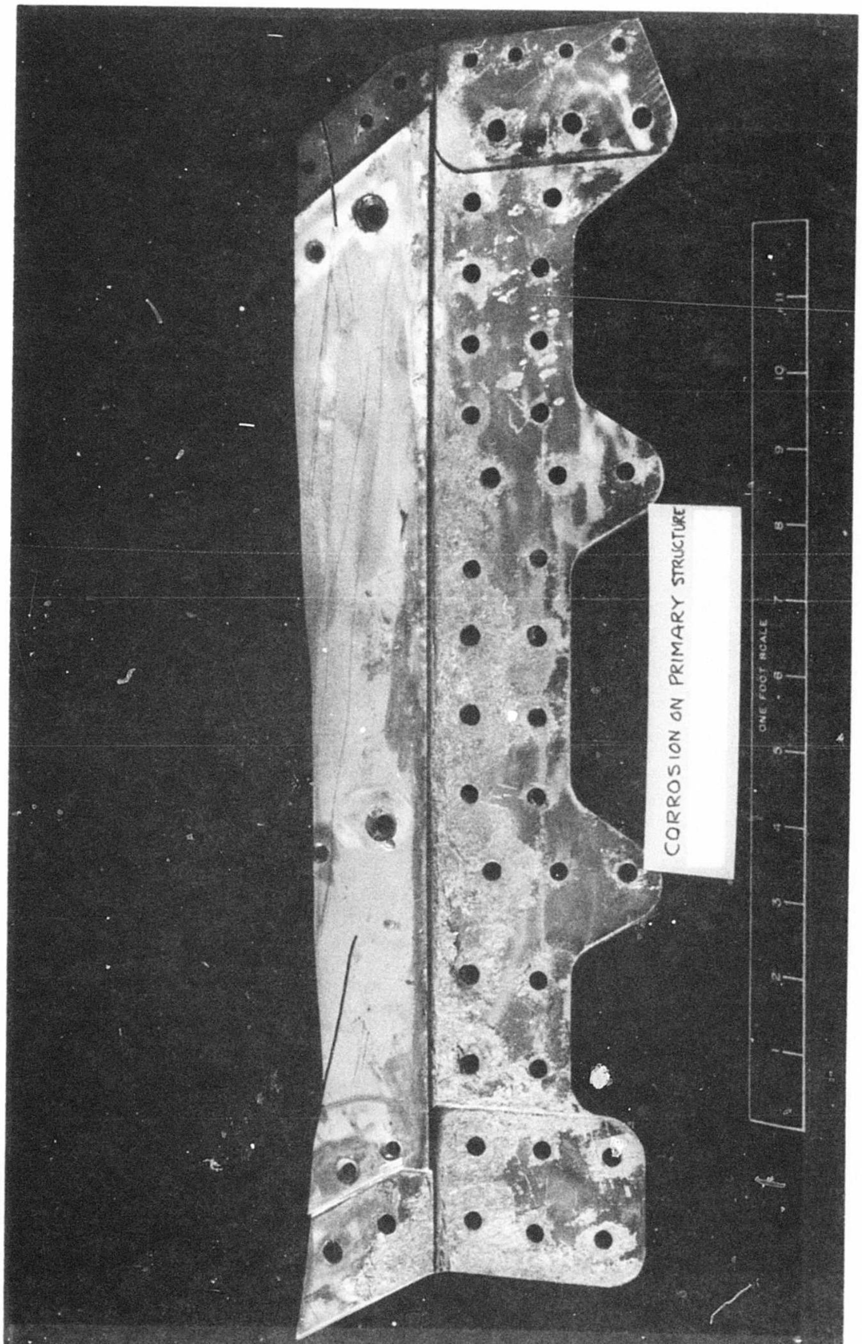


Figure 9. Corrosion of Stabilator Lower Skin

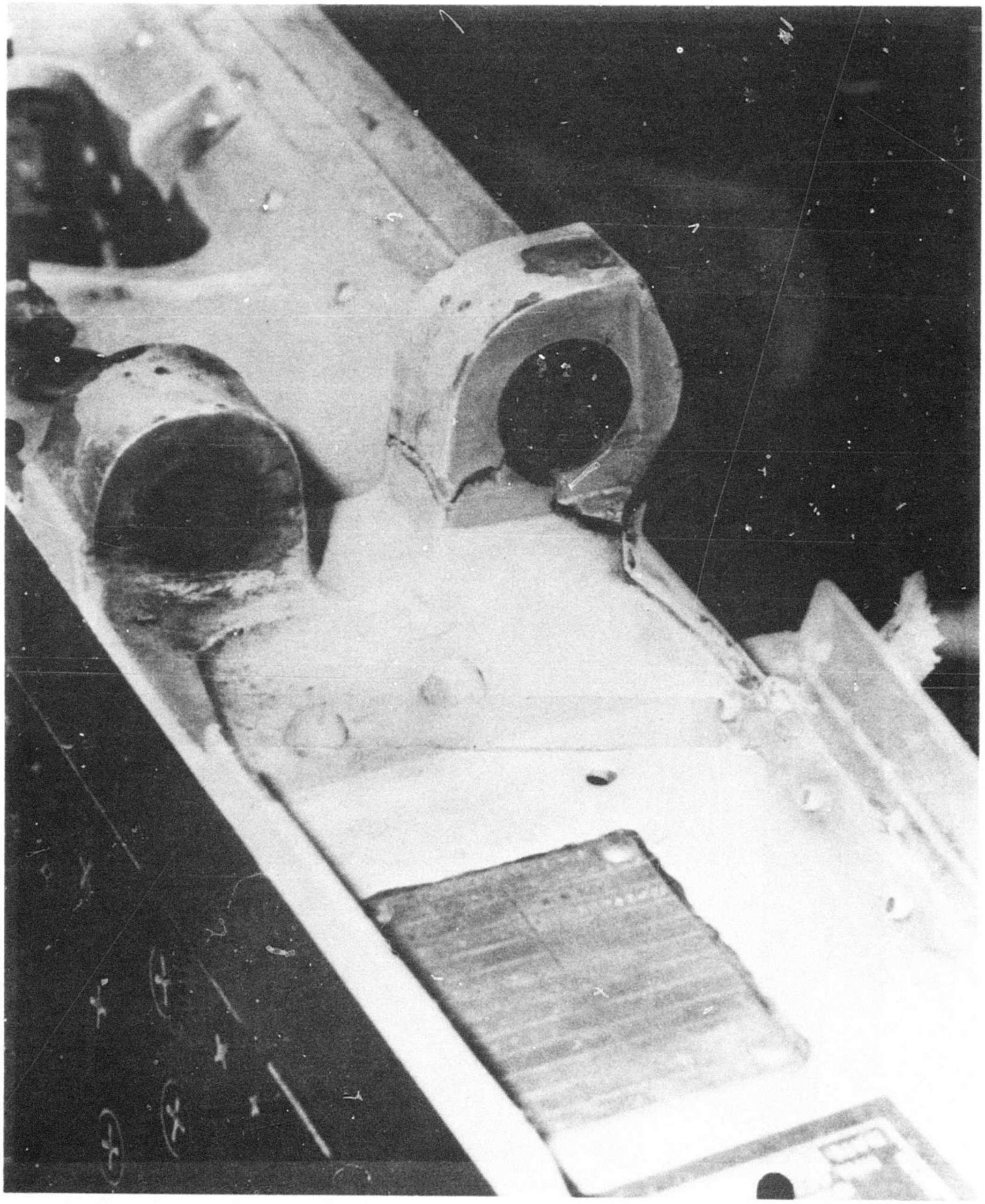


Figure 10. Failure of Wingtip Fold Rib Lower Aft Locking Lug

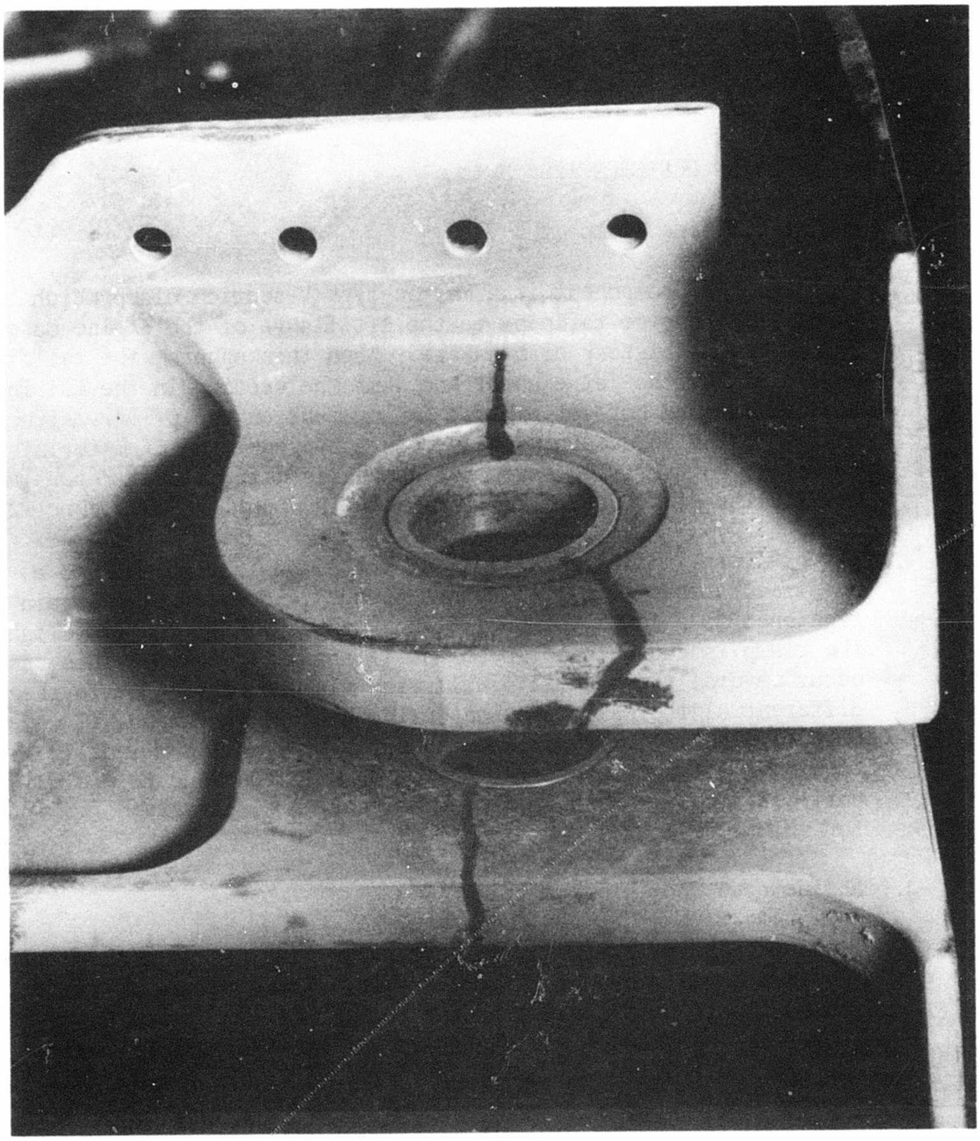


Figure 11. Cracks in Rear Spar

Cause: Stress corrosion cracking due to characteristics of 7078-T6 material.

3. Wing attach fitting

Problem: Cracks are frequently found around bushing area in a wing attach fitting.

Cause: Stress corrosion cracking caused by press fit bushing installation.

4.2.4 DETAIL DESIGN DEFICIENCIES

1. Engine tailpipe clamp

Problem: The component is a Marman-type V-section clamp which couples the engine tailpipe to the aft flange of the engine case. It has a long history of breakage. When this occurs, the tailpipe is forced aft by the jet exhaust and jams the opening in the fairing aft end causing hot jet exhaust impingement on the primary structure. Damage to the structure usually occurs by reduction of material strength; however, the extent of damage is difficult to determine. Usually the structure is replaced when in doubt. (See figure 12.)

Cause: Upon initial installation, the clamp is tightened to a specified torque (100 to 110 inch-pounds), loosened, then again tightened to a lower torque (25 inch-pounds). After a few flights, the clamp is found to be loose and is again tightened. This can occur several times until finally the clamp breaks. Due to the differential thermal expansion during heatup, the clamp, if completely tight, will experience some yielding. If continually tightened after this occurs, the clamp will continue to yield until it finally breaks. (Refer to section 7, paragraph 7.3.4 for solution.)

2. Engine cowl door hinge fitting

Problem: The powerplant cowl door hinge fittings at each end are experiencing structural failure. This failure occurs in the small radius flange area closest to the cowl door. The crack on the cowl hinge fitting extends diagonally across the inner fillet radius for approximately 3/16" and is readily visible to the naked eye. (See figure 13.)

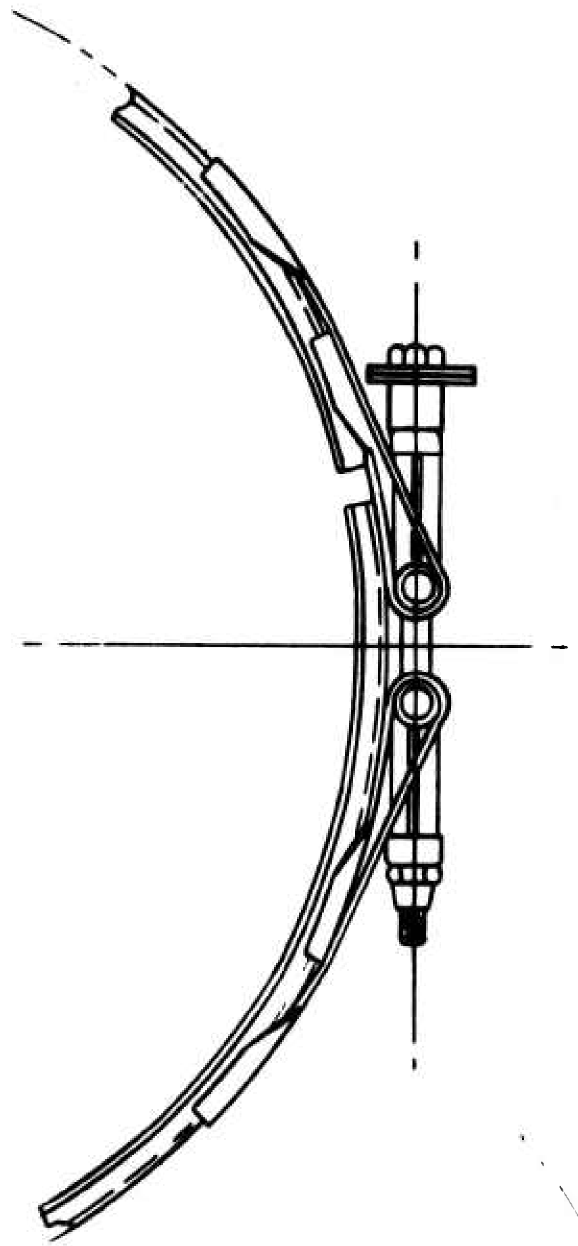


Figure 12. Tail Pipe Clamp Installation

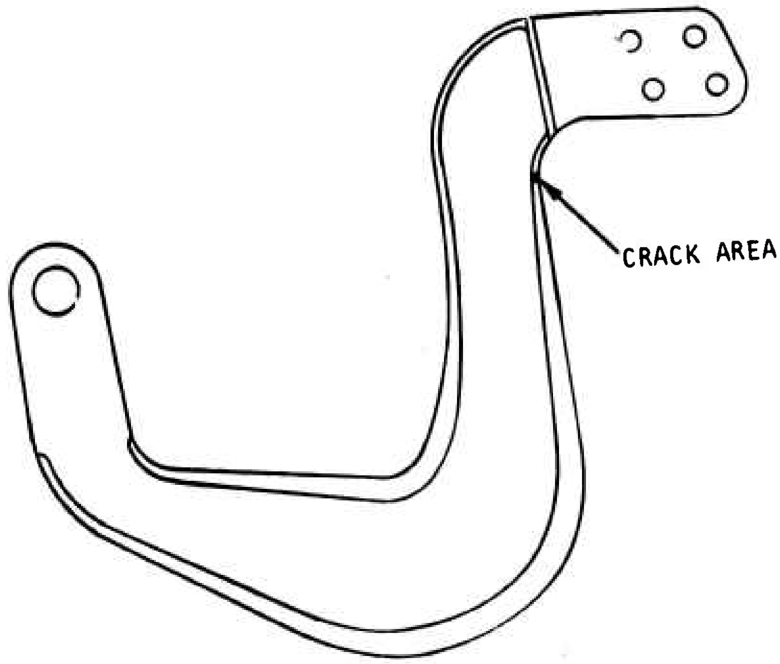


Figure 13. Engine Cowl Door Hinge

Cause: One of the more apparent causes of the cracked fitting is overextending the cowl door during servicing. This overextension allows the adjacent cowl structure to bottom out against the hinge fitting, thereby inducing bending stresses capable of failing the area in question. Another factor capable of causing the cracked hinge fitting failure is excessive loads induced by wind gust when the door is in its open position. A tension load in the hinge fittii may also result from the open position strut geometry. section 7, paragraph 7.3.5 for solution.)

3. Wing slat actuator doors

   Problem: High maintenance man-hours are being expended at operational bases on the doors in replacing failed parts in the mechanism and trying to keep the doors adjusted for proper alignment with the opening in the wing structure. Failures are occurring in the spherical head adjustment fittings due to a tension failure at the base of the head or by pulling the threaded insert out of the slat track fitting into which they attach. (See figure 14.)

   Cause: The cause of these failures has been analyzed as excessive load induced during slat retraction when the door is misaligned and catches on the wing structure before the slat has completed its retraction. Misalignment can occur from the following:

   a. Initial maladjustment

   b. Airload deflection

   c. Excessive play in the mechanism

   d. Slippage of the ball joint fittings relative to the door, as a result of loose bolts which normally clamp them to the door through a slotted hole and a friction pad. (Refer to section 7, paragraph 7.3.6 for solution.)

4. Nose gear single drag link

   Problem: Failure of a nose landing gear single drag link.

   Cause: Insufficient strength margin for hard landings and off-runway operations. With no backup link, the nose gear collapses causing extensive structural damage.

5. Wing skins - stress concentration

   Problem: Cracks have been found in the upper and lower wing skins at the inboard end. (See figure 15.)

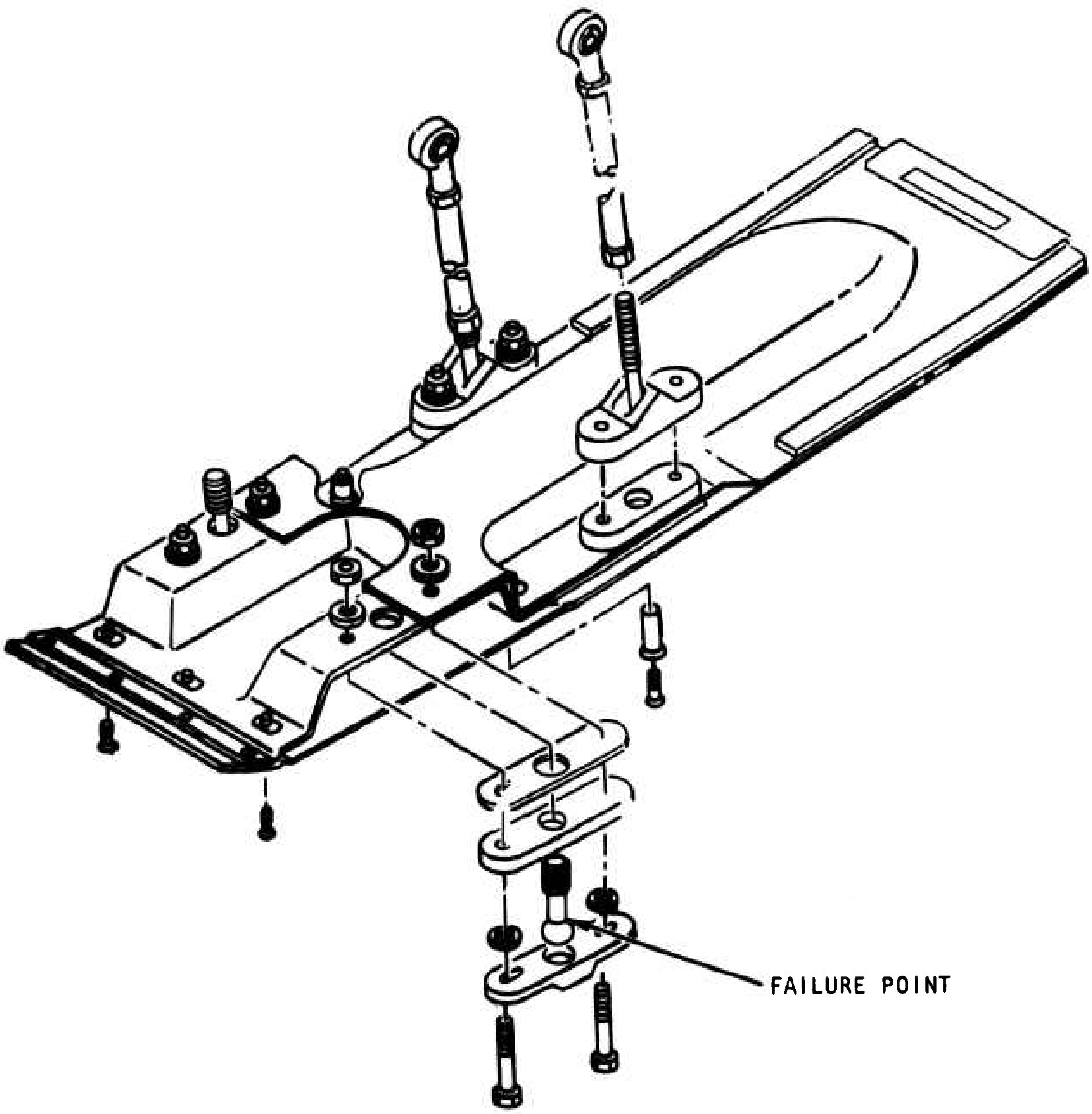


Figure 14. Typical Wing Slat Actuator Door Installation

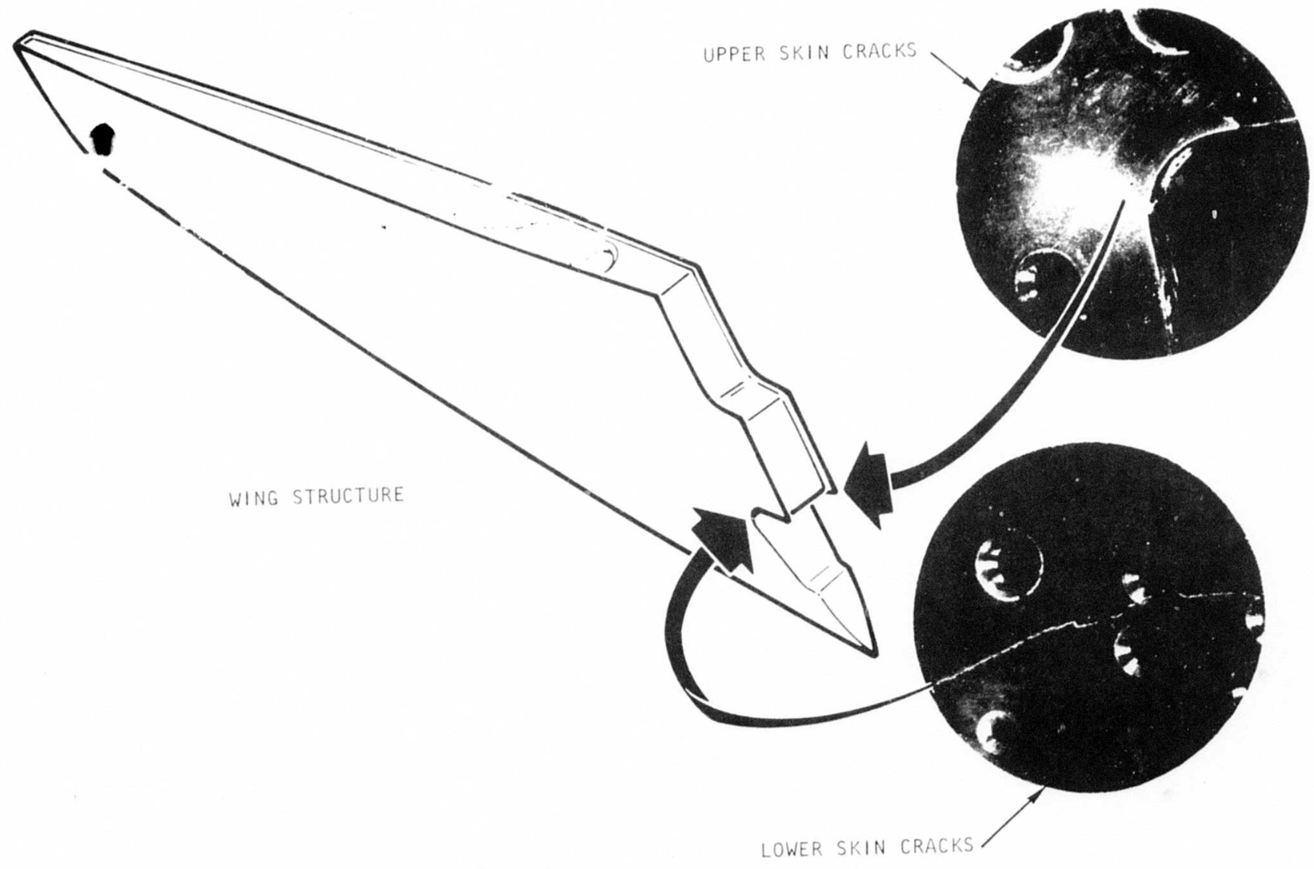


Figure 15. Cracks in Wing Skins

Cause: High stress concentration points caused by small inside corner radii.

6. Main landing gear uplock support

   Problem: Main landing gear door actuating linkage and uplock bumper are difficult to keep in rigged adjustment and requires frequent inspection and maintenance action.

   Cause: Linkage mounting bracket material is too thin, allowing bending and subsequent misalignment to occur.

7. Nose landing gear

   Problem: Nose landing gear on V/STOL aircraft experiences frequent failures to attach point to fuselage.

   Cause: Absence of a nose landing gear drag brace strut permits excessive loading of attach point.

8. Bonded skins - vertical and horizontal stabilizers

   Problem: Fiberglass skin on vertical stabilizer, and aluminum skin on horizontal stabilizer frequently separates from arrowhead shaped aluminum leading edge.

   Cause: Moisture enters joint between arrowhead shaped leading edge and skin, allowing debonding to occur.

9. One-piece sculptured metal skin

   Problem: One piece aluminum-fuselage sculptured skin found cracking at flanges.

   Cause: The root rib at the side of the fuselage has a compound curve (swarf) contour plus a dihedral break machined into its wide flanges. The spars lie on a chord percentage line and thus have constant bevels on their flanges. Mismatches in the root rib to the spar and skin when pulled together by fasteners result in considerable residual stresses being induced into the flanges which cause cracks to develop.

10. Windshield edge attach holes

    Problem: Windshield frequently found crazed and cracked at edges of attach holes.

Cause: Bird-resistant windshield made from polycarbonate, with no acrylic cladding or other protective coatings results in cracks from the edge attach holes, or crazing due to contaminants.

11. Fuselage access panel

Problem: Excessive man-hours required for removal and reinstallation on fuselage access panel. Panel is installed with 134 fasteners, of which there are 10 different types of fasteners and 31 different sizes.

Cause: Lack of adequate design considerations for maintenance and logistics.

4.2.5 FATIGUE DESIGN DEFICIENCIES

1. Nacelle aft cowl doors

Problem: An engine nacelle aft cowl door is experiencing structural failure of its longitudinal vane assembly and associated inner and outer cap angles. The failure is in the form of fatigue cracks in the vane assembly and in the corners of the cap angles. Damage in the form of delamination of the honeycomb core has also been found in the area of the outer vane assembly attach members.

Repair of the damage is tedious and, if extensive enough, requires special tooling to maintain door configuration during repair. Accessibility is the prime problem encountered in repairing the area. Replacement or repair of the vane assembly and caps from either end of the cowl door is limited by how far a man can reach into the openings. Repair beyond the end areas requires extensive door disassembly at a major repair depot.

The outer honeycomb panel is not removable and repairs must be effected from the inner surface. This is hampered by a lack of easily removable sections of structure to expose the damage. (See figure 16.)

Cause: Repeated cycling of the inner and outer door panels, due to airloads and engine sonic vibration, causes working of the vane assemblies and their inboard and outboard attach members resulting in fatigue cracks.

The constant tension load in the vanes also causes delamination of the honeycomb panel at the vane outer attach member, which is bonded integrally into the honeycomb panel. (Refer to section 7, paragraph 7.3.7 for solution.)

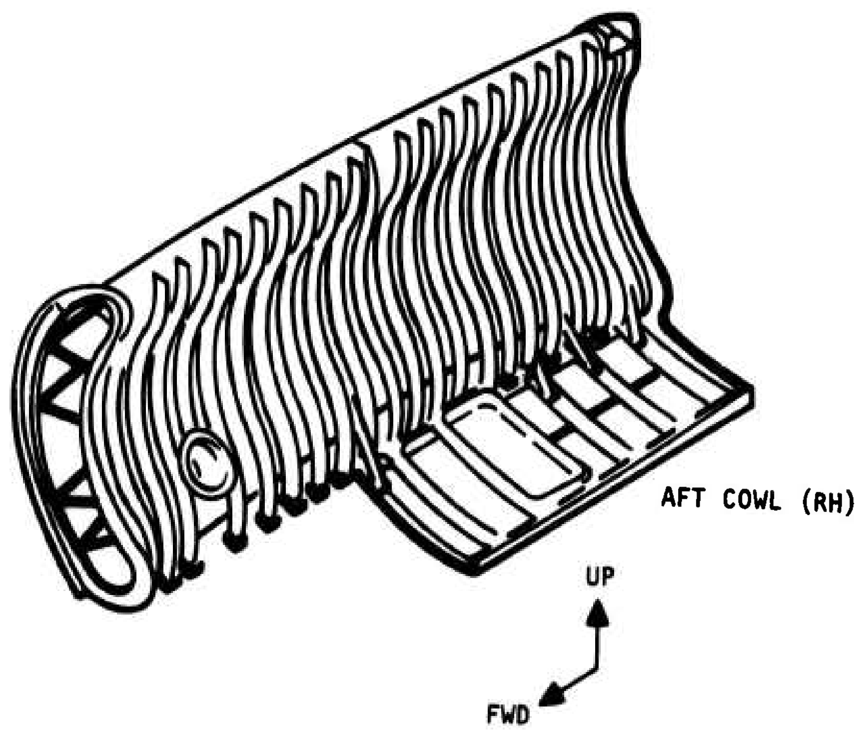


Figure 16. Engine Nacelle Aft Cowl Door

2. Fuselage fuel tank cavity liners

Problem: Fatigue cracks have developed in the aluminum floor and sidewall skins in four of the fuselage fuel tanks. The cracks tend to develop laterally along skin to frame riveted joints and propagate into the areas between the frames. (See figure 17.)

Cause: Fuel loads, due to surge, hydrostatic pressures, maneuvers, etc, in conjunction with shear and vibration loads from the engines, cause repeated cycling of the skins between frames until cracks start to form in aluminum material. (Refer to section 7, paragraph 7.3.8 for solution.)

3. Stabilator midspan joint rib

Problem: Fatigue failure of aluminum stabilator mid-span joint rib. (See figure 18.)

Cause: Rib receives fairly high temperatures from engine exhaust, and also a high frequency of stress cycles from aerodynamic and sonic vibration loads. The aluminum rib is attached to the inboard section of the stabilator constructed from titanium honeycomb, and to the outboard section made from aluminum honeycomb. The joint rib is subject to stresses from differences in materials, thermal expansion as well as bending loads, and also is subjected to moisture and corrosive gases.

4. Wingtip internal rib

Problem: Failures are experienced on a wingtip internal sheet metal rib. (See figure 19.)

Cause: Fatigue failure caused by omission of a stringer tie allows the skin to diaphragm under aerodynamic load and induce bending fatigue of rib flange.

5. Engine air inlet cowl web

Problem: Numerous cracks of an engine air inlet cowl web are being experienced. (See figure 20.)

Cause: Fatigue failure caused by sonic and engine vibration loads on sharp edge of chem milled beaded area.

Figure 17. Fuselage Fuel Cavity Liner Skins

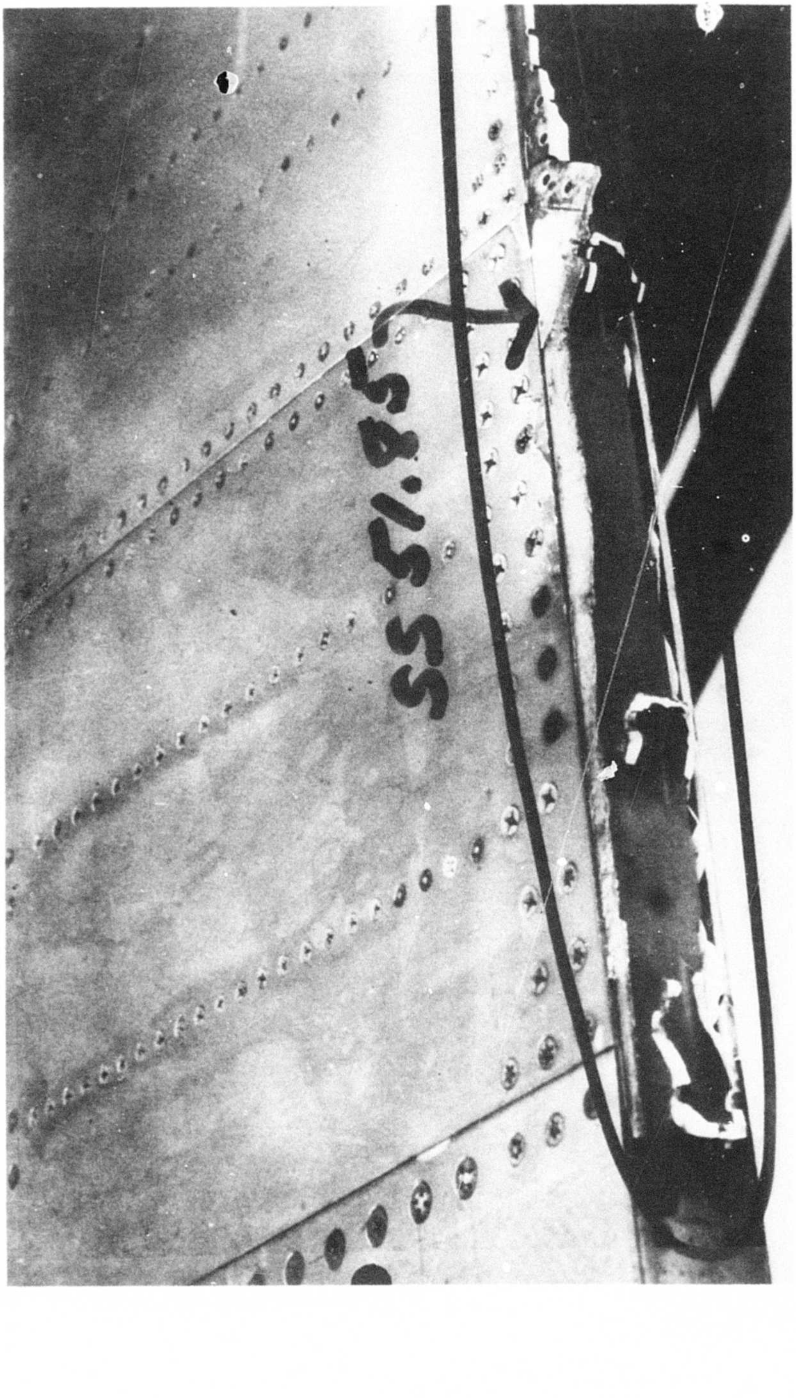


Figure 18. Fatigue Failure of Stabilator Rib

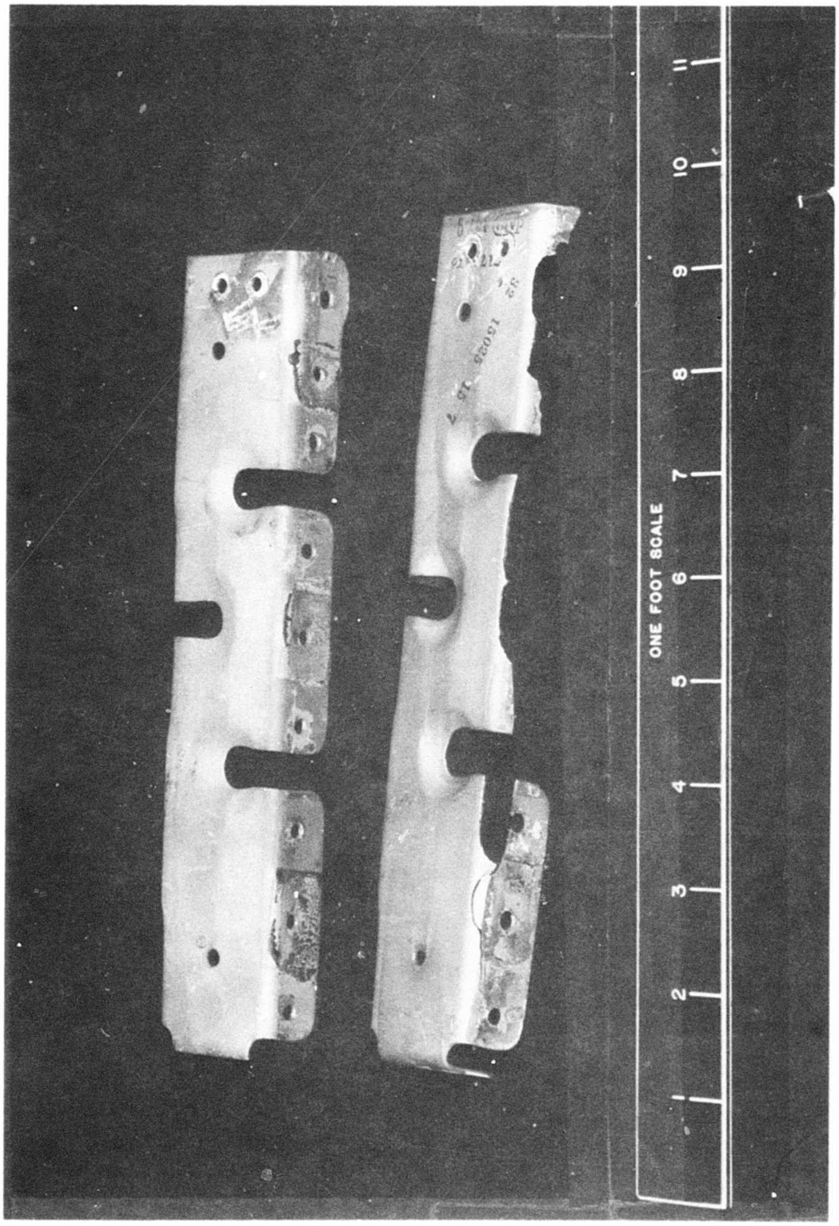


Figure 19. Fatigue Failure of Wingtip Internal Rib

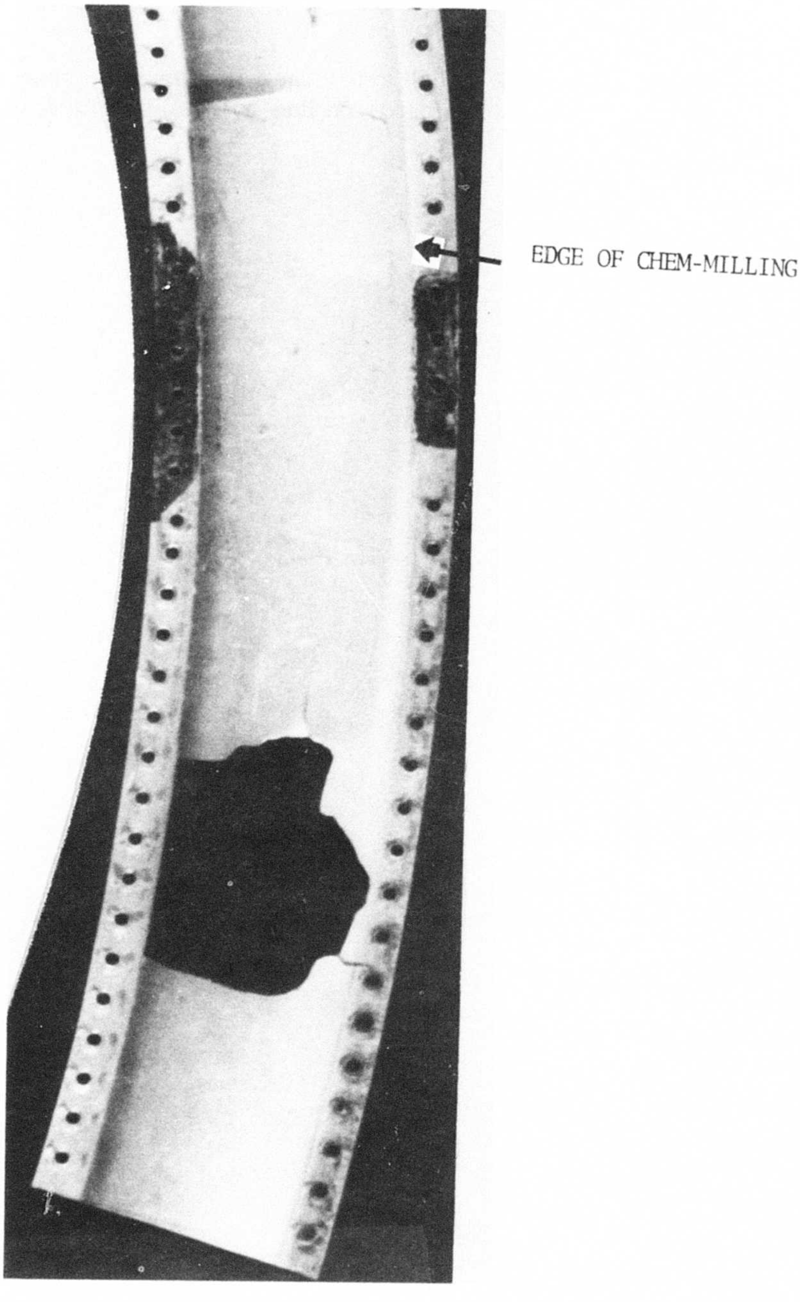


Figure 20. Cracks in Engine Air Inlet Cowl Web

6. Main landing gear wheel axle

Problem: Main landing gear wheel axles are found cracked. (See figure 21.)

Cause: Type 4130 steel axle cracks on upper surface near inboard bearing from stress corrosion due to residual tensile stresses (up to 50K).

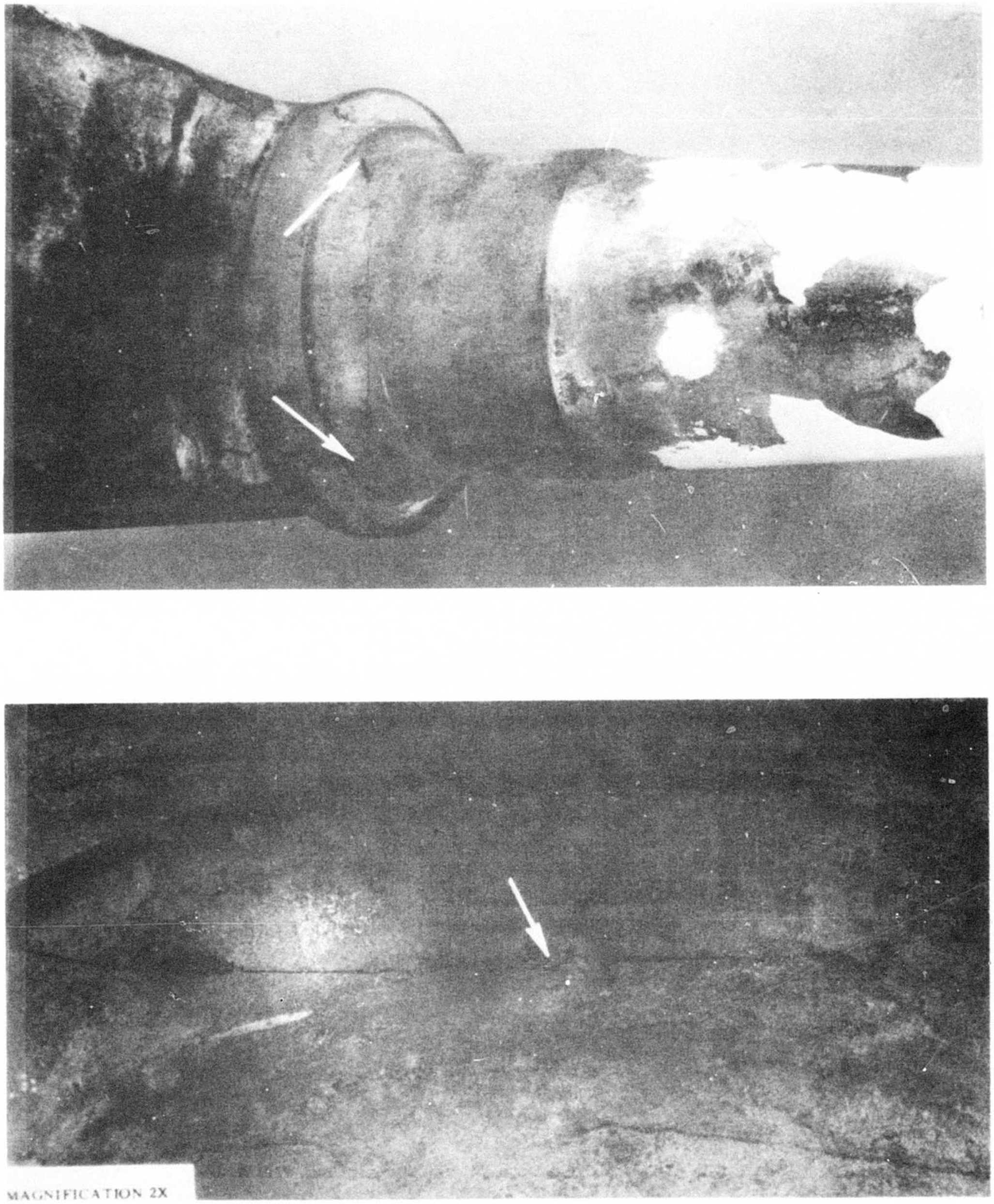


Figure 21. Crack in MLG Wheel Axle

5.0 GENERAL DESIGN CONSIDERATIONS

5.1 INTRODUCTION

This section contains general information on the candidate design techniques for low-cost structural maintenance features that may be incorporated into new aircraft systems. They represent only those techniques identified during the research conducted for the development of this design handbook and are presented as examples of some of the means to minimize maintenance costs. They are intended to provide designers with an appreciation of the means to reduce the cost of ownership to the Air Force. It is the responsibility of the structural designer to evaluate the potential problems for each specific aircraft system and to apply sound design judgment in the selection of the most effective combination of design features. The basic areas of consideration are contained in the following paragraphs.

5.2 MATERIAL SELECTION

Use commonly available materials where possible such as type 321 or 17-4PH CRES and 2024 aluminum alloys that have good fracture toughness characteristics and fatigue resistance.

5.3 STRUCTURAL ASSEMBLY ARRANGEMENTS

• Production breaks	Use mechanical joints, and limit sections to shippable sizes.
• Field breaks	Use removable-type fasteners, and locate joint for convenience of removal and replacement in service.

5.4 AIRCRAFT SUBSYSTEM INTERFACES

1. Provide Teflon-coated rub strips where movable control surfaces seal against the body.

2. Vibration damping material, which resists moisture, should be used on thin control surface skins to prevent fatigue. Skin on one side of flight control surfaces should be removable for inspection and repair of internal parts.

3. The landing gear wheel well areas and doors are extremely vulnerable to splashed mud and water. All exposed areas should be well protected to prevent corrosion, and all closed compartment doors should be sealed to prevent moisture entry.

4. Special care should be taken in landing gear wheel and strut wells to protect all areas from moisture and corrosive materials used on runways for snow and ice removal.

5. Give special consideration to sheet metal around gun bays and engine cowls where high vibration promotes fatigue cracking.

6. Armament gun bays should be so located that gun clip and case exits will not permit entry of ejected material into the engine inlets, thus becoming foreign object damage (FOD).

7. Assure adequate purging of corrosive gun gases in the gun bays and ammunition storage areas.

5.5 REMOVABLE AND HINGED DOORS, RADOMES, AND ENGINE COWLS

1. Design fastener holes sufficiently oversize to prevent loading the door under 1G static condition and to permit their easy ground removal and re-installation.

2. Provide hold-open devices on hinged doors which restrain the panel from racking and failing hinges.

3. Fastener hole patterns should be tooled for interchangeability. Provide, as a minimum, a weather seal of cured-in-place sealant with adequate land all around access door jams on external surfaces. Provide adequate access doors in structure for inspection and repair, and for possible future equipment installation modifications.

4. Provide adequate adjustment for alignment of latches on hinged cowl panels and doors.

5.6 CREW, PASSENGER, AND CARGO PROVISIONS

1. Provide noncorrosive pans around relief tubes, urinals, lavatories, and galley to prevent spilled liquids from contacting structure. Provide holders for coffee cups in crew area, and locate where spillage will not contact electrical equipment.

2. Soundproofing and insulating materials below floors should be separated from structural walls with standoffs to permit circulation of air and prevent moisture retention.

3. Provide hot air ventilation to areas below the cabin floor to keep the structure dry and free from moisture retention.

5.7 SEALING AND ORGANIC BONDING

1. Seal all exterior skin joints with sealing compound to prevent moisture entry and subsequent corrosion.

2. Sealing compound should be one which remains flexible and does not crack or shrink.

3. Use sealants in fuel bays which do not regress to a liquid state from prolonged exposure to fuel or environmental variations.

4. All exposed honeycomb core edges should be sealed.

5. Structural joining by organic bonding should not be used in low sump areas where moisture can collect. Cracks in the bonding can trap moisture and permit corrosion to occur.

6. Assure that organic bonding material used can withstand expansion and contraction of structural material joined, without cracking or disbonding.

5.8 CORROSION PREVENTION

1. Paint joining areas of all detail parts prior to assembly; then paint all areas after assembly with polyurethane paint.

2. Assure good ventilation of compartments in lower extremeties of structure to prevent continuous moist environment.

3. Assure good drainage paths of all compartments not sealed for pressure or fluids.

4. Prevent insulating material from contacting metal skins where condensation can be soaked up.

5. Where dissimilar metals are joined, an insulating layer of non-wicking material should separate them to prevent fretting and corrosion conditions.

6. Do not use phenolic impregnated fabric materials as exterior surfaces where moisture can be wicked to metal surfaces and cause corrosion.

5.9 FASTENERS

• Use standard types	Avoid special types were possible.
• Limit variations of sizes	Use one size and length on removable panels.
• Quick acting fasteners	Use stud retention type with self-aligning nut receptacles.
• Taper-Lok	Use only where other fasteners cannot meet fatigue requirements.

5.10 SPECIAL DESIGN CONSIDERATIONS

• Compartmentation	Provide access to all compartments for periodic inspection and repair.
• Panel size	Limit bonded sandwich panel size to 4 x 10 ft for repairability in existing autoclaves.
• Duct and shaft removal	Provide for removal of system ducts and shafts for repair or replacement without major disassembly of structure.
• Cast or forged parts	Make from sheet metal, where possible, for repairability.
• Transparencies	Design for interchangeability and easy replacement, especially windshields.

Design all fittings and high stressed parts for removal and replacement without having to remove other major structural elements or fabricate special replacement parts.

6.0 DETAIL DESIGN CONSIDERATIONS

6.1 CONSTRUCTION CONCEPTS

6.1.1 SKIN-STRINGER CONSTRUCTION

- Rib or frame flanges - Attach to stringer with a clip, and allow sufficient room in the flange width and rivit spacing from bend for installation of a repair angle.
- Stringer clips - Use at least one gage heavier than rib or frame. Provide access for replacement.
- Cutouts for stringers - Provide 60° by 3/8 flange around cutout unsupported areas.

6.1.2 SCULPTURED PLATE

- Allow generous fillet radii at base of upstanding stringers to prevent cracking.
- Allow sufficient stringer height and skin pad width for repair of cracks.
- Avoid sharp steps in thickness of skin.
- Allow generous radii in plate edge trim, where change in trim resulting in reintrant angle greater than 30° is required.

6.1.3 SANDWICH PANELS

- Keep panel size within capability of depot repair facilities (not over 4 by 10 feet). Seal complete panel to prevent moisture from entering and freezing, which disbonds face sheets.
- Close out and seal all exposed edges to prevent corrosion.
- Avoid sharp steps where face sheet thickness changes occur.

6.1.4 FORGINGS

- Avoid sharp machined fillet radii.
- Avoid stress corrosion prone materials.

- Design forgings to be replaceable without disassembly of other major structure.

- Avoid thin attachment flanges.

- Avoid pocket recesses which form moisture traps in installed position, or provide moistureproof seals to protect recesses.

6.1.5 CASTINGS

- Do not use for primary structure.

- Avoid thin webs and flanges.

- Avoid sharp fillet radii.

- Do not use where inspection access is unavailable.

- Assure drainage of all pockets and wells in installed position where complete sealing is not practical.

6.1.6 TRANSPARENCIES

- Refer to MIL-HDBK-17 for specific fabrication and installation requirements.

- Design transparent panels as floating units where loading permits. That is, edges should be flexibly mounted in a metal framework, which in turn can be bolted to the structure.

- Monolithic glass panels should have a shatter-resistant laminate on the side facing the crew.

- Stretched acrylic or polycarbonate plastic laminated panels should have an abrasion-resistant coating on the inner face, and either an abrasion-resistant coating or a thin glass facing on the outer face.

- Mount all transparencies so that they can be easily replaced.

- Clearly and permanently identify all transparent assemblies as to part number and manufacturer.

- Avoid passing mounting fasteners through transparencies, where possible.

- Seal all around outside edges of transparencies with a flexible sealant to prevent moisture entrapment in joints, which may cause corrosion.
- Locate all heating element wiring and connections where they cannot be accidentally broken or damaged.
- On heated panels, spare sensors should be incorporated into the laminate for use in case of primary sensor failure.

6.1.7 HINGED ATTACHMENTS

- Make all hinges and hinge fittings replaceable.
- Provide bushings which can be replaced after wear.
- Provide for removal of pins on piano-type hinges.
- Provide for lubrication where high operating loads exist.
- Assure adequate strength capability in hinges to withstand high wind loads or overtravel when opening doors.
- Do not make hinge fittings integral with large structural members such as spar caps or stringers where hinge damage would require structural replacement.

6.1.8 FASTENERS

Considerable controversy exists within the military and industry as to the choice of fasteners for structural use. Also, applications vary somewhat among different aircraft; however, some categorization can be made. The data given here are the result of queries made to maintenance personnel at all facilities visited.

Application	Recommendation
Medium-strength permanent joints	2024-T4 Al alloy rivets (<200° F) 2219-T81 Al alloy rivets (<325° F)
High-strength permanent joints	A286 CRES rivets 6-4 Ti bimetal rivets

Application	Recommendation
Blind rivets	CHERRYLOCK rivets NAS1398 and 1399 (high temp)
Infrequent removal panels	NAS1580 Hi-Torque bolts
Frequent removal panels	TRIDAIR panel fastener with hex recess

The fasteners which appear to require unduly high maintenance are:

- Taper-Loks - Difficult to remove for inspection and replace, especially aluminum type
- Jo-Bolts - Pin loosens, and fastener corrodes easily.
- Cam-Loc - Corrodes in receptacle
- Milson - Heads fail due to prestress. Retaining washers come off, and stud is lost. Holes are elongated by tapered ramp.

  (NOTE: This is not a deficiency of the fastener but the result of poor panel hole alignment.)

6.1.9 STRUCTURAL FASTENER SELECTION

Table I contains preferred parts selected for use in structural applications. The table provides an integrated summary of the characteristics of most structural fasteners in common usage.

The weight and cost information presented in the table may be used as a guide in selecting optimum parts. The terms low, medium, and high are intended for comparison to similar parts (i.e., rivets compared to rivets, etc). Where no direct comparison is possible, the terms refer to the relative difference between the comparative fastener and solid rivets.

TABLE I

AIRFRAME STRUCTURAL FASTENERS LISTING

BOLTS								
PART NO.	TYPE	MATERIAL AND STRENGTH	WEIGHT	COST	COST INSTALLED	MAXIMUM MISALIGN-MENT	MAX TEMP (°F)	REMARKS
NAS1580A	100° FLUSH TENSION HEAD HI-TORQUE RECESS.	STEEL 95 KSI SS	HIGH	LOW	HIGH	1/2°	450	LONG THREAD LENGTH FOR SHEAR APPLICATIONS REQUIRING SOME TENSION. USE TITANIUM FOR WEIGHT CRITICAL STRUCTURE. A-286 BOLTS ARE FOR ELEVATED TEMP APPLICATIONS. NAS1580A NOT TO BE USED IN TITANIUM.
NAS1580V		6-4 Ti 95 KSI SS	LOW	MED	HIGH		500	
NAS1580C		A-286 95 KSI SS	HIGH	MED	HIGH		1200	
NAS1581A	100° FLUSH SHEAR HEAD HI-TORQUE RECESS	STEEL 95 KSI SS	HIGH	LOW	HIGH	1/2°	450	LONG THREAD LENGTH FOR JOINTS PRIMARILY LOADED IN SHEAR. USE A-286 BOLTS FOR ELEVATED TEMP APPLICATIONS ONLY. USE LIMITED TO 1/4 AND LARGER. USE NAS1291C NUT WITH 95 KSI FASTENERS. NOT TO BE USED FOR DOOR APPLICATIONS.
NAS1581V		6-4 Ti 95 KSI SS	LOW	MED	HIGH		500	
NAS1581C		A-286 95 KSI SS	HIGH	MED	HIGH		1200	
NAS1578C	PAN HD HI-TORQUE RECESS	A-286 95 KSI SS	HIGH	MED	HIGH	1/2°	1200	PRIMARILY FOR SHEAR LOADING. A-286 BOLTS FOR ELEVATED TEMP ONLY. WHERE TOOL ACCESSIBILITY ALLOWS, USE NAS673V, HEX HEAD
NAS1578V		6-4 Ti 95 KSI SS	LOW	MED	HIGH		500	
NAS673V SERIES	HEX HEAD	6-4 Ti 95 KSI SS	LOW	MED	HIGH	1/2°	500	PRIMARILY FOR SHEAR LOADING.
NAS1303 SERIES		ALLOY STEEL 95 KSI SS	HIGH	LOW	HIGH		450	NOT FOR USE IN TITANIUM. USE NAS673V FOR WEIGHT CRITICAL STRUCTURE.

TABLE I

AIRFRAME STRUCTURAL FASTENERS LISTING (CONTINUED)

TAPER-LOK SHEAR BOLTS (SUPPLIER CODE IDENT 85495)

PART NO.	TYPE	MATERIAL AND STRENGTH	WEIGHT	MATING NUT PART NO.	MAX NUT MISALIGN-MENT	COST INSTALLED	MAX TEMP (°F)	REMARKS
TLV100	100° SHEAR HEAD	6-4 Ti 95 KSI SS	LOW	TLN1021CPD1 TLN1001CPD1	6° 1/2°	HIGH	500	(1) FOR FATIGUE CRITICAL INSTALLATIONS, WHERE OTHER LESS COSTLY TECHNIQUES WILL NOT MEET AIR VEHICLE LIFE REQUIREMENTS. (2) USE "D1" LUBE WITH CETYL ALCOHOL. CALLOUT CODE IS "E". (3) WHEN TLN1021 & TLN1024 NUTS ARE SELECTED, ADD ADDITIONAL GRIP LENGTH TO BOLTS AS REQUIRED PER PART DRAWING. (4) OVERSIZE FASTENERS ARE AVAILABLE FOR REWORK APPLICATIONS.
TLD100	100° SHEAR HEAD	PH13-8Mo 125 KSI SS	HIGH	TLN1021CPD1 TLN1001CPD1	6° 1/2°	HIGH	700	
TLV300L	100° TEN HEAD	6-4 Ti 95 KSI SS	LOW	TLN1010CPD1 TLN1021CPD1	1/2° 6°	HIGH	500	
TLD300L		PH13-8-Mo 125 KSI SS	HIGH	TLN1010DPD1 TLN1010PPD1 TLN1024DPD1	1/2° 1/2° 6°	HIGH	700	
TLV200	PROTRUDING SHEAR HEAD	6-4 TI 95 KSI SS	LOW	TLN1001CPD1 TLN1021CPD1	1/2° 6°	HIGH	500	
TLD200		PH13-8Mo 125 KSI SS	HIGH	TLN1001CPD1 TLN1021CPD1	1/2° 6°	HIGH	700	
TLV400	PROTRUDING TENSION HEAD	6-4 Ti 95 KSI SS	LOW	TLN1010CPD1 TLN1010DPD1 TLN1021CPD1	1/2° 6°	HIGH	500	
TLD400		PH13-8Mo. 125 KSI SS	HIGH	TLN1010DPD1 TLN1010PPD1 TLN1024DPD1	1/2° 6°	HIGH	700	

TABLE I

AIRFRAME STRUCTURAL FASTENERS LISTING (CONTINUED)

TAPER-LOK NUTS (SUPPLIER CODE IDENT. 85495)								
PART NO.	TYPE	MATERIAL	WEIGHT	OTHER CHARAC-TERISTICS	COST INSTALLED	MAXIMUM MISALIGN-MENT	MAX TEMP (F)	REMARKS
TLN1001CPD1	12 POINT SHEAR	A-286	LOW		HIGH	1/2°	800	
TLN1010CPD1	12 POINT TENSION	A-286	HIGH			1/2°		
TLN1010DPD1	12 POINT TENSION	PH13-8Mo	HIGH			1/2°		
TLN1010PPD1	12 POINT TENSION	INCO 718	HIGH			1/2°		
TLN1021CPD1	12 POINT SHEAR	A-286	MED	SELF-ALIGNING		6°		SAME AS H49492 NUT.
TLN1024DPD1A	12 POINT TENSION	PH13-8Mo	HIGH	SELF-ALIGNING		6°		IDENTIFIED WITH WHITE DOT TO DISTINGUISH FROM TLN1021CPD1.

TABLE I

AIRFRAME STRUCTURAL FASTENERS LISTING (CONTINUED)

LOCKBOLTS

PART NO.	TYPE	MATERIAL AND STRENGTH	WEIGHT	MATING COLLAR BASIC PART NO.	COST INSTALLED	MAXIMUM COLLAR MISALIGN-MENT	MAX TEMP (°F)	REMARKS
NAS2005V THRU NAS2012V	PROTRUDING TEN HEAD	6-4 Ti 95 KSI SS	MED	1080-2AC	MED	5°	325	(1) PRELOAD CONTROLLED BY STEM BREAK-OFF AND COLLAR SWAGING. (2) PERMANENT INSTALLATIONS. (3) LOCKBOLTS PREFERRED WHERE TOOL CLEARANCE IS ADEQUATE; OTHERWISE USE HI-LOKS. (4) LOCKBOLT USAGE IS LIMITED TO 3/16 DIA AND LARGER FOR SHEAR FASTENERS & 5/32 AND LARGER FOR TENSION HEAD FASTENERS. (5) MONEL COLLAR, DC-M( )E, AVAIL-ABLE IN 3/16 AND 1/4 DIA ONLY.
				6DC-EU			700	
NAS2105V THRU NAS2112V	100° TEN HEAD			1080-2AC			325	
				6DC-EU			700	
NAS2406V THRU NAS2412V	PROTRUDING SHEAR HEAD		LOW	2DC-2AC		3°	325	
NAS2406V THRU NAS2408V				DC-M( )E			700	
NAS2506V THRU NAS2512V	100° SHEAR HEAD			2DC-2AC			325	
NAS2506V THRU NAS2508				DC-M( )E			700	

LOCKBOLT COLLARS (SUPPLIER CODE IDENT. 29666)

PART NO.	TYPE	MATERIAL	WEIGHT	COST INSTALLED	MAXIMUM MISALIGN-MENT	MAX TEMP (°F)	REMARKS
DC-M( )E	DOUBLE ENDED	MONEL	LOW	MED	3°	800	USE ON SHEAR BOLTS.
2DC-2AC		2219 A1	LOW			325	
6DC-EU		A-286	MED		5°	1200	USE ON TENSION BOLTS.
1080-2AC	SINGLE ENDED	2219 A1	MED			325	

SCREW, SET

MS1806-4	HEX RECESS	CRES 300 SERIES	LOW	LOW			USED TO CAP OFF SEAL-ANT INJECTION BOLT HOLES.

TABLE I

AIRFRAME STRUCTURAL FASTENERS LISTING (CONTINUED)

PART NO.	TYPE	MATERIAL AND STRENGTH	INSTALLATION INFORMATION	WEIGHT	COST INSTALLED	MAXIMUM MISALIGN-MENT	MAX TEMP (°F)	REMARKS
NUTS, SELF-LOCKING								
NAS1291C( )M	HEX LOW HEIGHT SHEAR	A-286 125 KSI		LOW	MED	1/2°	450	(1) LOW HEIGHT, LOW WEIGHT. (2) USE A-286 NUT WITH TITANIUM FASTENERS (WITHOUT SILVER PLATE). (3) CAD PLATED STEEL MUST NOT BE USED WITH TITANIUM.
MS21042		STEEL 160 KSI		LOW	LOW	1/2°	450	
NUTS, CASTELLATED								
AN310	PLAIN CASTELLATED	CRES PASS. 125 KSI	ST0101LB0002 SAFETY PER LA0101-019	MED	HIGH	1°	700	TO BE USED WITH 106796A AND 106738A BOLTS FOR INLET DUCT APPLICATIONS REQUIRING LOCK WIRING.

TABLE I

AIRFRAME STRUCTURAL FASTENERS LISTING (CONTINUED)

NUTS, PLATE NUTS

PART NO.	TYPE	MATERIAL AND STRENGTH	WEIGHT	COST INSTALLED	FLOAT (RADIALLY)	MAX TEMP (°F)	REMARKS
MS21072L	ONE-LUG MINI	A-286 125 KSI	MED	HIGH	NONE	450	(1) MINIATURE PLATE-NUTS NOT TO BE USED IN PRIMARY LOAD PATH MEMBERS THAT ARE FATIGUE CRITICAL. (2) PRIMARILY FOR SHEAR LOADING. (3) NOT TO BE USED IN FUEL II SEALANT APPLIED TO PLATE NUT. (4) TORQUE TO BE CONTROLLED WHEN USED WITH 160 KSI FASTENERS. (5) SUPPLIER CODE IDENT. 72962
MS21074L	CORNER MINI						
MS210761	TWO-LUG MINI				.020		
MS210871	ONE-LUG MINI SIDE BY SIDE				NONE		
RMLHA3280MB (5)	TWO-LUG DEEP C'BORE MINI				.020		
NS103625-( )-( )TB	CORNER C'BORE				.025		(1) SUPPLIER CODE IDENT. 72962 MAY BE USED IN PRIMARY LOAD PATH MEMBERS THAT ARE FATIGUE CRITICAL.
F11184 (1)	TWO-LUG						
F11186 (1)	ONE-LUG						
NS103623-( )-( )TB	TWO LUG DEEP C'BORE	A-286 160 KSI			.030		(1) USE WHEN SCREW LENGTHS MUST BE THE SAME.
NS103624-( )-( )TB	ONE-LUG DEEP C'BORE						
F11399 (1)	ONE-LUG SELF-ALIGNING	INCO 718 160 KSI			.010 AT 8°		(1) SUPPLIER CODE IDENT. 72962 MAXIMUM MISALIGNMENT IS .010 RADIALLY AT 8° TILT.
F11400 (1)	TWO-LUG SELF-ALIGNING						

TABLE I

AIRFRAME STRUCTURAL FASTENERS LISTING (CONTINUED)

PART NO.	TYPE	MATERIAL AND STRENGTH	WEIGHT	COST INSTALLED	FLOAT (RADIALLY)	MAX TEMP (°F)	REMARKS
NUTS, PLATE NUTS - CONTINUED							
NS103834	TWO LUG, DIAGONAL SPACING	A-286 160 KSI	MED	HIGH	.030	450	
F1934-01	TWO LUG EXTRA DEEP COUNTERBORE	A-286 125 KSI			.030	450	SUPPLIER CODE IDENT. 15653
NS103621	TWO LUG MINIATURE COUNTERBORE	A-286 125 KSI			.020	1000	(1) SILVER PLATED FOR HI-TEMP ONLY. (2) NOT TO BE USED ON TITANIUM.
NS103817	ONE LUG	A-286 125 KSI			.030	1200	
NS103804	TWO LUG	INCO 718 160 KSI			.030	1200	
NUTS, ANCHOR (SLIP SLIDE)							
A6293	TWO-LUG	STEEL 125 KSI	MED	HIGH	.190 DIA = .150R .250 DIA = .100R	450	(1) SLIP SLIDE FLOATING TYPE. (2) FOR TENSION-ONLY APPLICATIONS. (I.E., NON-STRUCTURAL FAIRING APPLICATIONS.) (3) NOT TO BE USED ON TITANIUM.
NUTS, DOME NUT PLATES							
NS103535E	TWO-LUG SPACER	A-286 160KSI	HIGH	HIGH	.025		(1) CODE FOR DRY FILM LUBE ONLY, DO NOT USE SILVER PLATE. (2) ON M14735 (a) TORQUE TO BE CONTROLLED WHEN USED WITH 160 KSI BOLTS. (b) DO NOT USE IN DESIGNS WHERE PRIMARY LOAD PATH MEMBERS ARE FATIGUE CRITICAL. (c) SUPPLIER CODE IDENT. 80539
NS103731E	ONE-LUG SPACER	A-286 160 KSI			.015		
M14735HE	TWO-LUG RED. RIV. SP	A-286 125 KSI			.015		
NS103606-( )-( )TB	CORNER	A-286 160 KSI			NONE		
NS103609-( )-( )TB	ONE-LUG SIDE BY SIDE COUNTERBORED	A-286 160 KSI			.020		

TABLE I

AIRFRAME STRUCTURAL FASTENERS LISTING (CONTINUED)

PART NO.	TYPE	MATERIAL AND STRENGTH	WEIGHT	COST INSTALLED	FLOAT (RADIALLY)	MAX TEMP (°F)	REMARKS
NUTS, CHANNEL (SUPPLIER CODE IDENT 72962)							
G11192J	REPLACEABLE NUT ELEMENT	A-286 NUT AL ALLOY CHANNEL	HIGH	HIGH	.015	265	FOR USE IN FUEL TANK AREAS.
G11370J							FOR USE IN NON-FUEL AREAS COUNTERBORED.
NUTS, BARREL (SUPPLIER CODE IDENT 97393)							
SL4001-( )M	BARREL	A-286 220 KSI SIZES -4 THRU -12 INCO 718 220 KSI SIZES -14 THRU -20	HIGH	HIGH	.028 TOTAL LENGTHWISE FLOAT	450	FOR USE WITH TITANIUM OR PH13-8Mo BOLTS.
SLR4001	RETAINER	INCO X-750					INDEX OR CLEARANCE TYPE.
INSERTS							
MS21209	HELICAL COIL	CRES	LOW	MED		800	STRENGTH BASED UPON THREAD SHEAR AREA OF PARENT MATERIAL.

TABLE I

AIRFRAME STRUCTURAL FASTENERS LISTING (CONTINUED)

RIVETS, CONVENTIONAL

PART NO.	TYPE	MATERIAL AND STRENGTH	WEIGHT	COST INSTALLED	MAXIMUM MISALIGN-MENT	MAX TEMP (°F)	REMARKS
MS20426DD	100°TEN	2024-14 41 KSI				200	3/16 DIA ONLY
MS20426AD	100°TEN	2117-T4					
MS20427M	100°TEN	MONEL 49 KSI		MED	ON UPSET HEAD	800	(1) SHEAR HEAD RIVET JOINT FATIGUE CHARACTERISTICS BETTER THAN FULL HEAD FLUSH RIVET (2) INSTALLATION LOADS (GUN DRIVEN) ARE HIGH AND SHOULD BE AVOIDED WHERE DRIVING CAN DAMAGE STRUCTURE. THE A-286 RIVETS REQUIRE THE HIGHEST DRIVING LOADS, MONEL THE LOWEST
NAS1200	100°SHEAR	A-286 90 KSI				1200	
NAS1199	100°TEN	A-286 90 KSI	HIGH	HIGH			
NAS1198	UNIV HEAD	A-286 90 KSI					
MS20470DD	UNIV HD	2024-14 42 KSI				200	3/16 DIA ONLY
MS20615M	UNIV HD	MONEL 49 KSI	LN	HIGH		800	SEE INSTALLATION NOTES FOR MS20427M

TABLE I

AIRFRAME STRUCTURAL FASTENERS LISTING (CONTINUED)

RIVETS, BI-METAL (SUPPLIER CODE IDENT. 11815)

PART NO.	TYPE	MATERIAL AND STRENGTH	WEIGHT	SUPERSEDED PART NUMBER	COST INSTALLED	MAXIMUM MISALIGN-MENT	MAX TEMP (°F)	REMARKS
CSR922	100° TEN	6-4 Ti 95 KSI	LOW	CSR912	MED	5° ON UPSET HEAD	500	THESE RIVETS HAVE A 6-4 Ti HEAD AND SHANK. THE TAIL IS COLUMBIUM ALLOY, AND IS THE ONLY PART UPSET DURING DRIVING. USE ONLY 5/32 and 3/16 DIA FOR IMPACT GUN INSTALLATIONS. MUST BE USED IN 1/32 GRIP INCREMENTS. USE IN PREFERENCE TO LOCKBOLTS OR HI-LOKS FOR SHEAR APPLICATIONS. BI-METALS ARE LOWER COST AND WEIGHT. NOT TO BE USED WITH OFFSET TOOLS.
CSR924	100° SHEAR			CSR914				
CSR925	UNIV HD			CSR915 (INACTIVE FOR FUTURE DESIGN)				

TABLE I

AIRFRAME STRUCTURAL FASTENERS LISTING (CONTINUED)

RIVETS, BLIND (5) SUPPLIER CODE IDENT. 11815

	PART NO.	TYPE	MATERIAL AND STRENGTH	WEIGHT	STEM TYPE	COST INSTALLED	MAXIMUM MISALIGN-MENT	MAX TEMP (°F)	REMARKS
	NAS1339D	100° TEN	2017	LOW	LOCKED SPINDLE	MED	5° ON BLIND SIDE	200	
	NAS1399M	100° TEN	MONEL	HIGH				900	
	NAS1399C	100° TEN	A-286	HIGH				1200	A-286 SOFT STEM
(5)	CR2664	100° SHEAR	A-286					1200	A-286 SOFT STEM. NOT RECOMMENDED WHEN SKIN THICKNESS ALLOWS USE OF FULL HD TYPE (NAS 1339C)
(5)	CR2A38	100°	2219-T42	LOW	BULB			325	① ②
(5)	CR2A62	100°	2219-T42	LOW	LOCKED SPINDLE			325	
(5)	CR2A64	100° SHEAR	2219-T42	LOW				325	NOT RECOMMENDED WHEN SKIN THICKNESS ALLOWS THE USE OF FULL HEAD TYPE (i.e., CR2A62)
(5)	CR2838	100°	INCONEL 600	MED	BULB			1200	③ ④
	NAS1398D	UNIV HD	2017	LOW	LOCKED SPINDLE			200	
	NAS1398M	UNIV HD	MONEL	HIGH				900	
	NAS1398C	UNIV HD	A-286	HIGH				1200	
(5)	CR2A63	UNIV HD	2219-T42	LOW				325	NAS1398 CONFIGURATION IN 2219-T42 MATERIAL
(5)	CR2A39	UNIV HD	2219-T42	LOW	BULBED			325	① ②
(5)	CR2839	UNIV HD	INCONEL 600	MED				1200	FOR HIGH TEMP APPLICATIONS. FOR BLIND SIDE SHEET THICKNESSES BELOW MIN FOR NAS1398. ③ ④

TABLE I

AIRFRAME STRUCTURAL FASTENERS LISTING (CONTINUED)

RIVETS, BLIND (CONTINUED) (SUPPLIER CODE IDENT. 11815)

PART NO.	TYPE	MATERIAL AND STRENGTH	WEIGHT	STEM TYPE	COST INSTALLED	MAXIMUM MISALIGN-MENT	MAX TEMP (°F)	REMARKS
CCR264CS-3	100°	A-286	MED	PULL THRU			400	FOR PLATE NUTS ONLY (5)
CR2643	PROTRUDING HD	A-286 95 KSI SS	MED	LOCKED SPINDLE			1200	(1) HARD STEM. A-286 (2) MAY BE USED IN ALUMINUM WITH WET PRIMER (3) AVOID THE USE OF FLUSH SHEAR HEAD WHENEVER POSSIBLE
CR2644	100° SHEAR HEAD							

NOTES:

1. FOR APPLICATIONS WHERE BLIND SIDE SHEET THICKNESS IS BELOW MINIMUM ACCEPTABLE FOR CR2A62, CR2A63, OR CR2A64.
2. USE IN PLACE OF CR2A62 WHEN A HIGHER STRENGTH IS REQUIRED FOR ACOUSTIC ENVIRONMENT.
3. FOR USE IN ENGINE INLET REGIONS WHEN A BLIND FASTENER IS REQUIRED & ACOUSTIC ENVIRONMENTS.
4. FOR LOWER STRENGTH REQUIREMENTS, USE CR2A38 OR CR2A39.
5. NOT TO BE USED FOR STRUCTURAL JOINTS. TO BE USED ONLY WHEN SOLID RIVETS CANNOT BE INSTALLED IN NUT PLATES. PULL THRU STEM RIVETS NOT TO BE USED WHERE SEALING IS REQUIRED.

TABLE I

AIRFRAME STRUCTURAL FASTENERS LISTING (CONTINUED)

PINS, HI-LOK (SUPPLIER CODE 73197, 92215, 56878)

PART NO.	TYPE	MATERIAL AND STRENGTH	MATING COLLAR	WEIGHT	COST INSTALLED	MAX COLLAR MISALIGN-MENT	MAX TEMP (°F)	REMARKS
HL10V	PROTR SHEAR HEAD	6-4Ti 95 KSI SS	HL70	LOW	MED	0°	325	(1) SHEAR HEAD PINS ARE PRIMARILY FOR SHEAR LOADING.
			HL82-( )ADW			7°		(2) TENSION HEAD PINS ARE FOR SHEAR APPLICATIONS WHERE SOME SECONDARY TENSILE CAPABILITY IS REQUIRED.
			HL94DU			0°	700	(3) USE PH13-8Mo ONLY WHEN 95 KSI SHEAR STRENGTH DOES NOT MEET REQUIREMENTS.
HL11V	100° SHEAR HEAD		HL70			0°	325	(4) USE AS ALTERNATE FOR LOCKBOLTS WHERE TOOL CLEARANCE IS INADEQUATE
			HL82-( )ADW			7°		(5) SERVICE TEMPERATURE MAY BE LIMITED BY COLLAR
			HL94DU			0°	700	(6) ALUMINUM COLLAR MAY NOT BE USED WITH PH13-8Mo PINS
HL12V	PROTR TEN HEAD		HL73DU			0°	600	(7) 6-4Ti HI-LOK USAGE IS LIMITED TO DIAMETERS
			HL89DU-( )ADW			7°		(8) NOT TO BE USED IN ENGINE INLET AREA WHERE LOOSENED PARTS CAN BE INGESTED INTO ENGINE
HL13V	100°TEN HEAD		HL73DU			0°		(9) COLLARS ARE NOT REUSABLE THREADED AND CONTROLLED TORQUE.
			HL89DU-( )ADW			7°		(10) USE ONLY THE PIN COLLAR COMBINATIONS SHOWN.
HL40	PROTR SHEAR HEAD	A-286 95 KSI SS	HL94DU	MED		0°	700	(11) PERMANENT INSTALLATION COLLAR SHEAR-OFF CONTROLS PRELOAD.
			HL175DU-( )ADW			7°		
			HL97DU			0°	1200	
			HL185DU-( )ADW			7°		
HL41	100° SHEAR HEAD		HL94DU			0°	700	
			HL175DU-( )ADW			7°		
			HL97DU			0°		
			HL185DU-( )ADW			7°	1200	
HI48	PROTR TEN HEAD		HL78			0°		
HL49	100° TEN HEAD		HL78			0°		
HL644LL	PRIOR SHEAR HEAD	PH13-8Mo 125 KSI SS	HL94DU			0°	600	
			HL175DU-( )ADW			7°		
HL645LL	100° SHEAR HEAD		HL94DU			0°		
			HL175DU-( )ADW			7°		
HL646LL	PROTR TEN HEAD		HL73DU			0°		
			HL89DU-( )ADW			7°		
HL647LL	100° TEN HEAD		HL73DU			0°		
			HL89DU-( )ADW			7°		

TABLE I

AIRFRAME STRUCTURAL FASTENERS LISTING (CONTINUED)

COLLARS, HI-LOK (SUPPLIER CODE IDENT. 73197, 92215, 56878)

PART NO.	TYPE	MATERIAL	WEIGHT	STRENGTH	COST INSTALLED	LUBRICANT	MISALIGN-MENT	MAX TEMP (°F)	REMARKS
HL 70	SHEAR	2024-T6	LOW	DETER-MINED BY PIN STRENGTH	MED	DRY FILM LUBE	0°	325	(1) NON-REUSABLE, THREADED, CONTROLLED TORQUE, COLLARS. (2) USE ALUMINUM COLLARS ON 6-4 Ti PINS ONLY. (3) A-286 COLLARS TO BE USED ON A-286 PINS ONLY.
HL73DU	TENSION	17-4 CRES	MED					600	
HL82-( )ADW	SHEAR SELF-ALIGN	2024-T6	LOW				7°	325	
HL89DU-( )ADW	TENS SELF-ALIGN	17-4 CRES	MED					600	
HL94DU	SHEAR	303 CRES					0°	700	
HL97DU	SHEAR	A-286						1200	
HL175DU-( )ADW	SHEAR SELF-ALIGN	303 CRES					7°	700	
HL185DU-( )ADW	SHEAR SELF-ALIGN	A-286						1200	
HL78	TENSION	A-286				SILVER PL	0°	1200	

TABLE I

AIRFRAME STRUCTURAL FASTENERS LISTING (CONTINUED)

FASTENERS, QUICK-ACTING STUDS (SUPPLIER CODE IDENT. 29372)

PART NO.	TYPE	MATERIAL AND STRENGTH	WEIGHT	MATING RECEP-TACLES	COST INSTALLED	MAX TEMP (°F)	REMARKS
CA17055	100° SHEAR	STEEL 95 KSI	HIGH	CA17057 CA17058	HIGH	265 265	USE CA17056 RET RING ①
CA17078	100° SHEAR	A-286 95 KSI		CA17070		550	USE CA1753 RET RING ②
CA17089	PROTRUD-ING HD	STEEL 95 KSI		CA17057 CA17058		265 265	USE CA17056 RET RING ①
CA17083	PROTRUD-ING HD	A-286 95 KSI		CA17070		550	USE CA1753 RET RING
				CA17014T		1000	② HIGH TEMP APPLICATIONS ONLY. USE CA17104T RET RING

RECEPTACLES, RINGS, AND WASHERS (SUPPLIER CODE IDENT. 29772)

PART NO.	TYPE	MATERIAL	WEIGHT	COST INSTALLED	MAXIMUM MISALIGN-MENT	FLOAT (RADIALLY)	MAX TEMP (F)	REMARKS
CA17014T	TWO LUG RIVET MOUNTED RECEP-TACLE	A286, SIL.PL.	HIGH	HIGH	1/2°	.030	1000	① USE ONLY WHEN SEALING IS NOT REQUIRED.
CA17057		ALUM.& STEEL					265	
CA17058		CRES, STEEL, & ALUM.				.025	265	① USE ONLY WHEN SEALING IS REQUIRED.
CA17070		CRES PASSI-VATE					550	USE ONLY WHEN SEALING IS REQUIRED. ②
CA17056	RETAIN-ING RING	CRES CAD. PLATE					450	USE WITH CA17085 OR CA17089 STUDS ① INSTALL RINGS WITH CA1786-T11 TOOL
CA1753	RETAIN-ING RING	CRES PASSI-VATE					550	② INSTALL RINGS WITH CA1786-T11 TOOL
CA17104T		INCO718					1200	USE WITH CA17083 OR CA17078 STUDS.
CA17095	DIMPLED WASHER	CRES PPASSI-VATE	LOW	LOW			700	USE WITH CA17055 OR CA17078 STUDS IN SOFT MATERIAL ONLY.

NOTES: ① NOT TO BE USED IN TITANIUM. ② FOR USE IN TITANIUM.

TABLE I

AIRFRAME STRUCTURAL FASTENERS LISTING (CONTINUED)

FASTENERS, BLIND INT. THD. SELF-LOCKING							
PART NO.	TYPE	MATERIAL	WEIGHT	COST INSTALLED	MAX MISALIGN-MENT	MAX TEMP (°F)	REMARKS
NAS1669	PROTRUD-ING HD	STEEL CAD. PLATE	HIGH	HIGH	5° ON BLIND SIDE	450	SUPERSEDES PLT 210 (3) (1)
NAS1670	100° HEAD						SUPERSEDES PLT 110 (3) (1)
NAS1671	PROTRUD-ING HD	A-286				1200	SUPERSEDES PLT 220 (3) (2)
NAS1672	100° HEAD						SUPERSEDES PLT 120 (3) (2)

FASTENERS, BLIND PULL TYPE							
PART NO.	TYPE	MATERIAL AND STRENGTH	WEIGHT	COST INSTALLED	MAX MISALIGN-MENT	MAX TEMP (°F)	REMARKS
MS90353	100° TEN HEAD	STEEL CAD. PLATE	HIGH	HIGH	5° ON BLIND SIDE	450	(1) ONLY FOR BLIND BOLT USE IN ENGINE STRUCTURE WHERE FOD IS POSSIBLE TO ENGINE, FOR OTHER APPLICATIONS USE 10 BOLTS.

NOTES:
(1) NOT TO BE USED IN TITANIUM.
(2) FOR USE IN TITANIUM.
(3) NOT TO BE USED IN ENGINE INLET STRUCTURE. WHERE FOD IS POSSIBLE TO ENGINE USE MS90353.

TABLE I

AIRFRAME STRUCTURAL FASTENERS LISTING (CONTINUED)

PART NO.	TYPE	MATERIAL AND STRENGTH	WEIGHT	COST INSTALLED	NOMINAL CLEARANCE	MAX TEMP (°F)	REMARKS
WASHER							
MS35338	LOCK WASHER	STEEL	LOW	LOW	.013	450	USE IN COMPLIANCE WITH AND 10476.
		CRES	LOW	LOW		700	
NAS1598C( )Y	SEALING WASHER	CRES & RUBBER	MED	MED	-.010	275	PRESSURE SEALING FUEL RESISTANT
NAS1598D( )Y		7075-T6 & RUBBER	MED	MED		200	
AN960	PLAIN	STEEL	MED	LOW	.015	450	COMMON GRIP WASHER
AN975	RECESSED	STEEL	HIGH	HIGH	.015	450	USE WITH AN386 TAPER PINS

TABLE I

AIRFRAME STRUCTURAL FASTENERS LISTING (CONTINUED)

FASTENERS, BLIND INT. THD. SELF-LOCKING

PART NO.	TYPE	MATERIAL	WEIGHT	COST INSTALLED	MAX MISALIGN-MENT	MAX TEMP (°F)	REMARKS
NAS1669	PROTRUD-ING HD	STEEL CAD. PLATE	HIGH	HIGH	5° ON BLIND SIDE	450	SUPERSEDES PLT 210 ③ ①
NAS1670	100° HEAD						SUPERSEDES PLT 110 ③ ①
NAS1671	PROTRUD-ING HD	A-286				1200	SUPERSEDES PLT 220 ③ ②
NAS1672	100° HEAD						SUPERSEDES .LT 120 ③ ②

FASTENERS, BLIND PULL TYPE

PART NO.	TYPE	MATERIAL AND STRENGTH	WEIGHT	COST INSTALLED	MAX MISALIGN-MENT	MAX TEMP (°F)	REMARKS
MS90353	100° TEN HEAD	STEEL CAD. PLATE	HIGH	HIGH	5° ON BLIND SIDE	450	① ONLY FOR BLIND BOLT USE IN ENGINE STRUCTURE WHERE FOD IS POSSIBLE TO ENGINE, FOR OTHER APPLICATIONS USE 10 BOLTS.

NOTES: ① NOT TO BE USED IN TITANIUM.
② FOR USE IN TITANIUM.
③ NOT TO BE USED IN ENGINE INLET STRUCTURE. WHERE FOD IS POSSIBLE TO ENGINE USE MS90353.

6.1.10 BALLISTIC DAMAGE

Aircraft structure can be damaged and weakened by the primary or secondary damage caused by hostile weapon effects. The primary damage mechanism is penetration of the structure by a projectile or warhead fragment. The response of the structure is dependent upon the size, impact velocity, and attitude of the projectile, together with the structural material properties, construction, and operational stresses. For a straight-in penetration (normal to the surface) of ductile metal, such as aluminum alloy, the entry hole would be approximately the same size as the projectile. Figure 22 shows the penetration of a typical aluminum alloy skin by a .50 caliber projectile while under simulated operating loads. The damage is restricted to a clean hole with back surface "petaling" and no crack propagation. The exit damage in this type of construction is dependent upon the attitude (yaw angle) of the projectile and the type of intervening structure and/or components that are also penetrated. Figure 23 shows typical exit damage from a "tumbled" projectile.

The angle of projectile penetration (obliquity) can also influence the degree of damage sustained. Figure 24 shows an extreme case where the projectile path was nearly parallel with the structural skin, and a long path of damage was experienced.

More brittle materials, such as 7075-T6 aluminum, that might be used for high stress applications are susceptible to extensive damage from crack propagation . An example is shown in figure 25, where large triskelion cracks were experienced.

Transparencies used for aircraft structures are also susceptible to damage from small-arms fire. The extent of such damage is a function of the material properties, construction, and operating stresses. Cast and stretched acrylics tend to crack or shatter from impact, and large sections are sometimes torn out. Polycarbonates are more resistant to such damage propagation. Ballistic-resistant transparencies, such as bullet-resistant glass, are generally fabricated in layers with flexible adhesives between them. Such configurations tend to experience "spidering" cracks from ballistic impact and/or penetration. An example of this type of damage is shown in figure 26.

Sandwich structure, such as honeycomb material contained between face sheets, responds to ballistic impact generally as shown in figures 27 and 28. Deformation of the face sheets is typical of thin-skin materials. Delamination of the honeycomb core from the face sheets is also experienced. Figure 29 illustrates the level of damage encountered with low grazing angle hits in the honeycomb panel structure. Extensive delamination is experienced beyond the exit hole in addition to core damage from secondary particles.

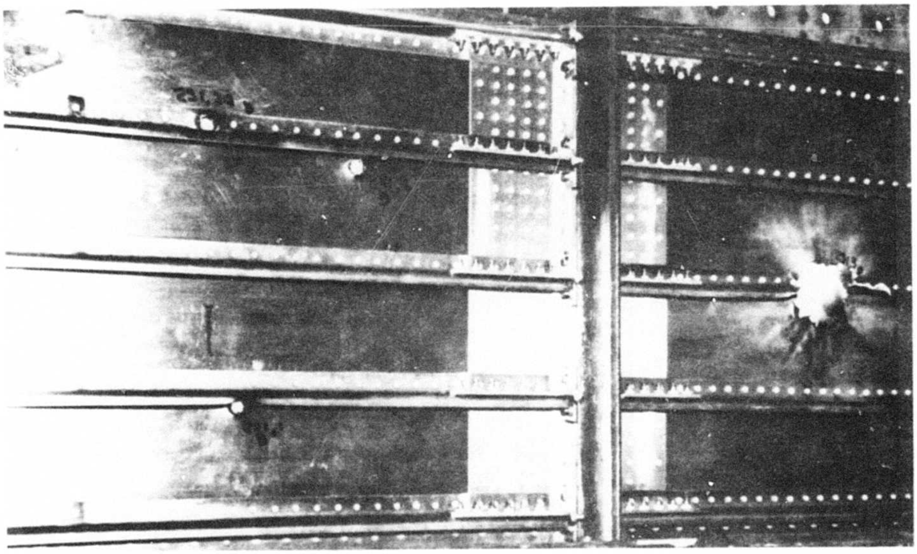


Figure 22. Aluminum Skin Projectile Damage - Normal Entry

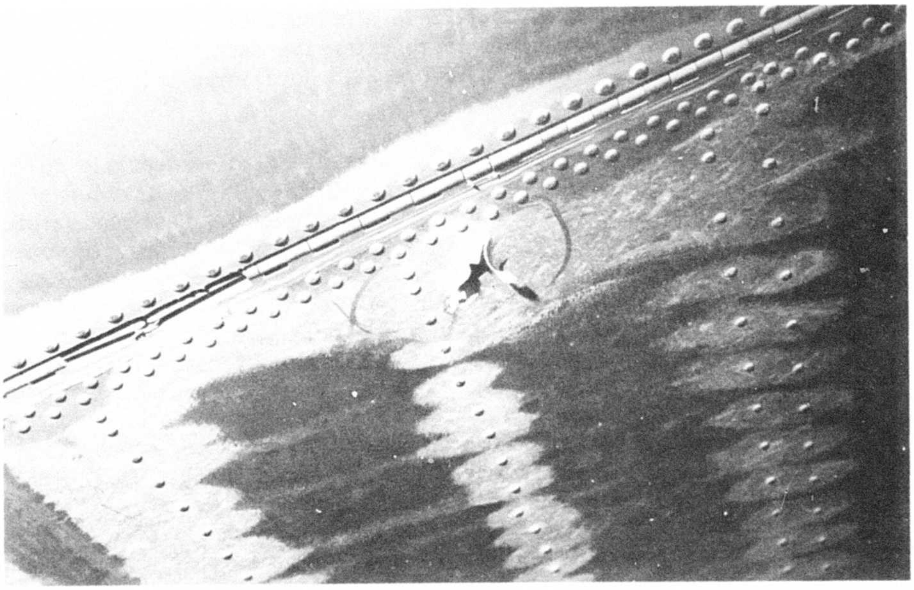


Figure 23. Tumbled Projectile Exit Damage

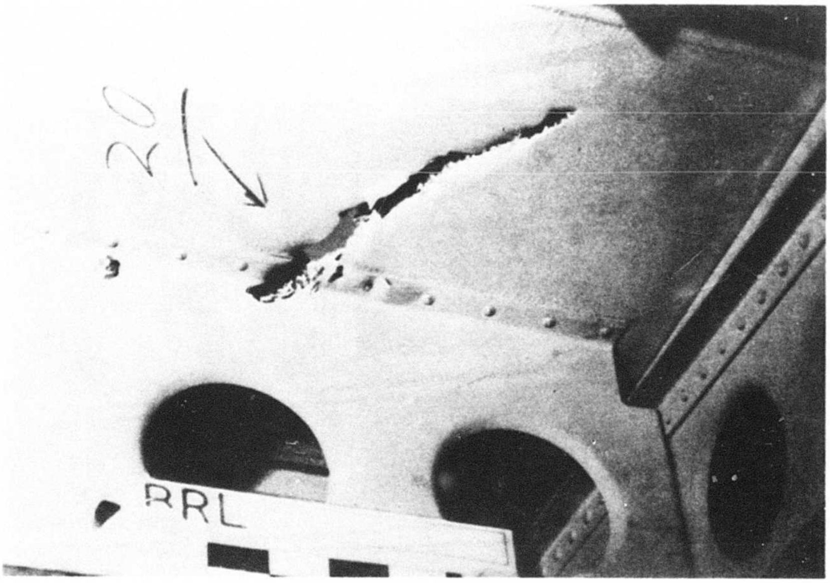


Figure 24. Structural Skin Damages (Projectile High Obliquity)

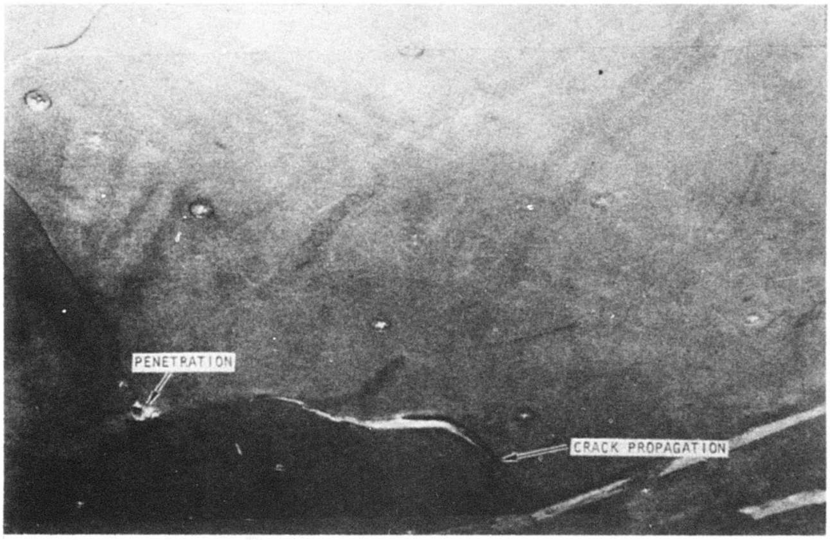


Figure 25. Brittle Material Ballistic Effect Crack Propagation

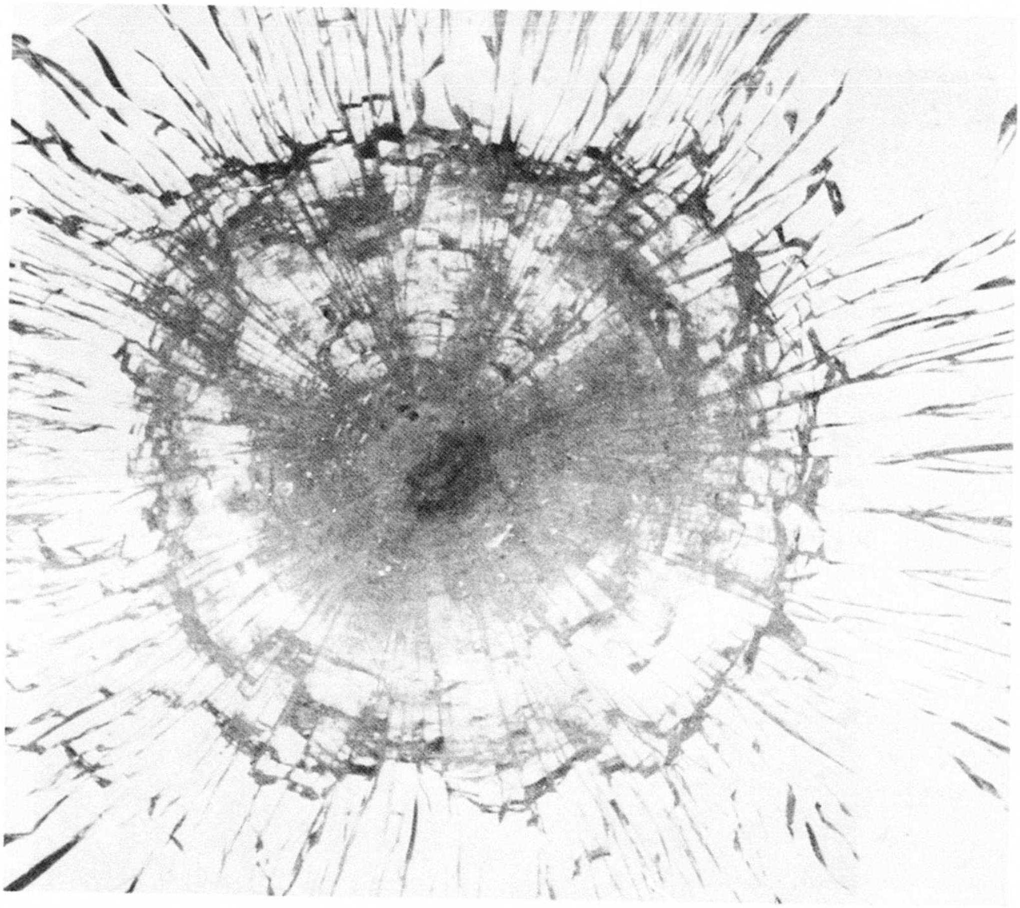


Figure 26. "Spidering" Effect Ballistic Impact Damage to Ballistic-Resistant Transparency Test Sample

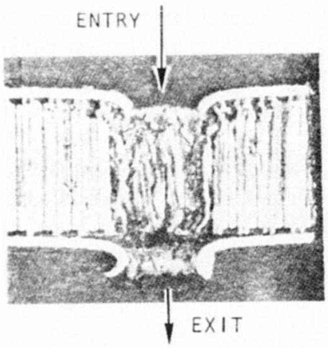


Figure 27. Honeycomb Panel Damage - 0°

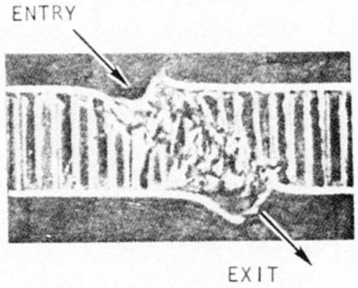


Figure 28. Honeycomb Panel Damage - 45° Obliquity

•CALIBER .50 BULLET AT GRAZING ANGLE OF 18 DEGREES PRODUCES EXTENSIVE DELAMINATION AND DAMAGE DUE TO SECONDARY PARTICLES.

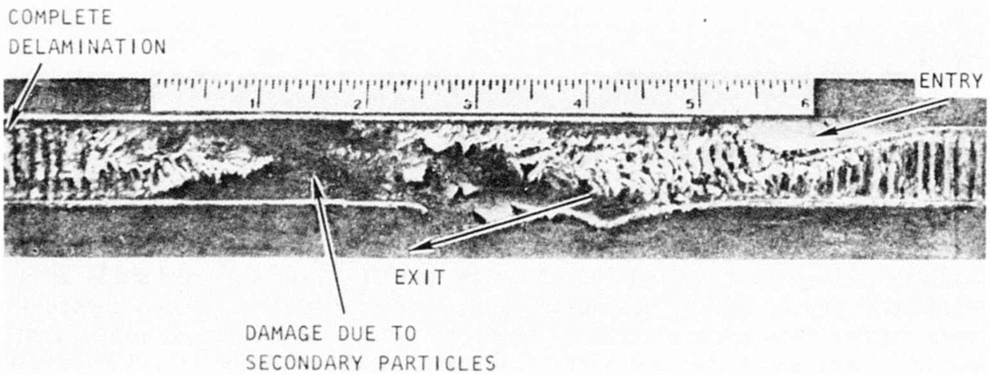


Figure 29. Honeycomb Panel Damage - 72° Obliquity

Secondary damage effects are those not directly caused by ballistic impact, but by initiation of a response or condition from another element within the air vehicle. Spallation and debris generated by a projectile penetration of structure may cause secondary damage to sensitive elements. Liberation and ignition of flammable fluids or vapors can cause fire or explosion conditions that can seriously damage the airframe. High thermal conditions may also be generated by ballistic damage to engine hot sections or hot air bleed lines. This can produce hot gas "torching" effects that may degrade the capability of the structure to adequately carry operational loads or cause distortions that, in turn, can affect the operational capability of other subsystems.

Projectile impact into structural sections containing liquids, such as fuel, creates a liquid pressure pulse or "hydraulic ram" effect that is transmitted directly to the supporting structure. Depending upon the size and kinetic impact energy of the projectile, cracking, bulging, or rupture of the structure can occur.

6.1.10.1 General Design Considerations

Generally, small-arms weapon damage to aircraft structure will not cause loss of the aircraft or appreciably affect its capability to fly. There are a number of factors that must be considered to insure that newly developed materials and construction concepts are employed in a manner that does not adversely affect survivability of the aircraft. The following are basic design factors that should be carefully considered in the initial design process where basic structural configurations are defined.

6.1.10.2 Material Selection

When selecting aircraft structural material, it is important to use those with good fracture-toughness qualities to prevent or minimize crack propagation from small-arms damage. The critical plane strain-stress intensity factor (K_{I_c}) is used in the field of fracture mechanics as a fracture index. Material type, heat-treat condition, and grain direction are variables that influence its capability to resist crack growth. Fracture mechanics is a relatively new technical discipline and, as such, must rely upon physical tests rather than pure analytical means to determine the crack resistance of specific structural designs after ballistic damage. Table II is a listing of some typical types of airframe construction materials with their typical and minimum fracture indexes (K_{I_c}) for longitudinal and transverse grain directions at room temperature. The values shown are for thousands of pounds per square inch stress (ksi) times the square root of the crack length ($\sqrt{l}$) in

Table II

ROOM-TEMPERATURE FRACTURE-TOUGHNESS PROPERTIES

Material	Condition	Typical K_{Ic} (KSI x $\sqrt{\ell}$)		Minimum K_{Ic} (KSI x $\sqrt{\ell}$)	
		L	T	L	T
2024 aluminum alloy	-T3X and -T4X	40	35	27	23
	-T6X and -T8X	25	19	17	13
7075 aluminum alloy	-T6X	24	21	17	14
	-T76X	27	25	20	18
	-T73	30	26	21	18
6Al-4V titanium	Cond A	60	60	40	40
	Diffusion bonded	75	75	50	50
9Ni-4Co-0.30C steel	220 ksi min ult	110	110	73	73
PH13-8Mo corrosion-resistant steel	H1000	90	90	60	60
	H1100	110	110	73	73

L = Longitudinal grain direction

T = Transverse grain direction

K_{Ic} = Critical plane-strain-stress intensity factor

ℓ = Crack length in inches

inches. For example, for 2024-T3X aluminum alloy material with a 1-inch crack, a longitudinal stress of over 40,000 psi will cause crack growth.

6.1.10.3 Basic Design Concepts

Three basic types of construction can be used for aircraft structures: thin skin/stringer, sandwich, and sculptured plate. The selection of

construction type for each major structural element (i.e., fuselage, wing, and empennage) must consider the type and level of damage by hostile small-arms weapon effects that can be tolerated.

Thin skin/stringer construction provides more ballistic damage tolerance than other types of construction when ductile, high-fracture-toughness materials are used. Multi-load path construction should be used to allow fail-safe response of the structure if damaged. Stringers, frames, and longerons with wide exposure area should be used in preference to those having their area concentrated in a narrow exposure which can cause a larger percentage of load-carrying capability to be lost when struck by a projectile. Attachments for the transfer of high concentrated loads should employ multiple fasteners and be designed for adequate strength following ballistic damage, to permit safe recovery of the aircraft under combat maneuvering conditions.

Sandwich construction consists of face sheets attached to a separating core of low density material. One of the most widely used types is bonded honeycomb. Fiber glass or plastic material laminates are examples of facing sheets that have been under recent development and use. Composites, such as boron filaments bonded with epoxy resins, have been under research and development for new aircraft structure, since they offer high strength-to-weight ratios that are needed for higher performance requirements. Graphite fibers are also under development as a new construction material. Selection of the basic material for sandwich construction must consider the strength remaining in the load-carrying elements when subjected to single- and multiple-projectile impacts.

Sculptured plate (integrally stiffened) structure is fabricated from one piece of material by mechanical, electrical, or chemical means. Material is removed to leave relatively thin walls integral with heavier stiffening lands and attachment sections. This type of construction is generally used for highly loaded panels, or "shell-like" applications. This type of construction should be used with discretion due to the potential danger of extensive crack propagation and the limited combat area repairability characteristics. Experience with sculptured plate construction has shown that small-arms fire damage has required the replacement of major structural elements of this type where such damage levels in skin/stringer and sandwich-type construction were easily repaired.

Careful selection of the basic material and its heat treatment is essential to obtain good fracture-toughness characteristics for the specific application that will prevent or minimize crack propagation.

6.1.10.4 Major Element Replacement

Extensive damage can be sustained by aircraft structures from the indirect effects of enemy gunfire. Forced or crash landings can be caused by damage to flight-essential subsystems. Consideration should be given to design concepts that will permit easy removal and replacement of major structural elements to facilitate rapid combat area repair and return of the aircraft to operational status. Interchangeability of major structural sections should also be considered to permit rapid repair by cannibalization of damaged aircraft.

6.1.10.5 Detail Design Considerations

The majority of aircraft structural design considerations for survivability enhancement against small-arms weapon effects are applicable to almost all portions of the airframe. These considerations apply to three types of construction: thin skin/stringer, sandwich, and sculptured plate. Each has been used for fuselage, wing, and empennage construction.

6.1.10.6 Thin Skin/Stringer Construction

The following techniques for minimizing the consequences of small-arms projectile impacts on thin skin/stringer type structures should be considered:

- Select materials with high fracture-toughness values to minimize or prevent crack propagation following ballistic damage.
- Consider the use of bonded "doublers" on high-strength stressed skin panels, such as 7075-T6 aluminum alloy, that may be susceptible to catastrophic failure from a single hit by a small-arms projectile. A thin layer "strap" of fiber glass can be bonded in proximity to the skin to provide such protection. A crack can be arrested by placing a number of fibers across a given zone of stress normal to the line of expected crack direction, thus reducing the stress intensity below the level required to propagate the crack. Test programs have shown that a significant improvement in crack arrest can be achieved for very modest penalties. Figure 30 illustrates placement of thin fiber glass tape on a typical high-stress panel to provide a crack-arrest feature.

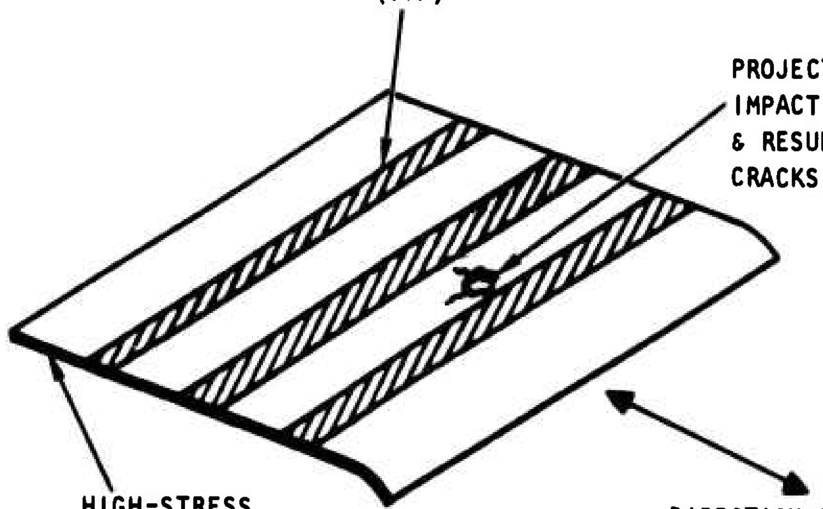


Figure 30. Crack-Arrest Straps.

6.1.10.7 Sandwich Construction

The following techniques for minimizing damage from small-arms projectile impacts on sandwich construction should be considered:

- Provide high-strength face sheet to inner core bonding material in areas where fuel or other liquids are carried to prevent or minimize delamination from liquid pressure pulse (hydraulic ram) effects caused by ballistic impacts.

- Consider the use of "planking" construction techniques to limit face sheet delamination from core material as a result of projectile impacts.

- Use high-temperature-tolerant bonding materials in areas where short-term fires or high-temperature air can be experienced from small-arms damage, to minimize loss of structural integrity.

6.1.10.8 Sculptured Plate Construction

Where sculptured plate construction is used, the following design techniques should be considered:

- Use materials with high fracture-toughness characteristics to resist crack propagation from ballistic damage. For example, use 2024-T4 aluminum alloy in place of higher strength, but more brittle, 7075-T6 aluminum alloy. Select heat-treat condition of material to obtain good fracture-toughness values.

- Use "planking" construction for structural areas primarily under tension-type loads to limit crack propagation from battle damage.

- Avoid straight lines of fasteners over large sections subject to high stress loads to limit rapid "zippering" effects due to projectile damage.

- Design sculptured sections with large radii, and avoid abrupt changes in sections where ballistic impact energies can develop high stress concentrations.

6.1.11 BEARING APPLICATION/ATTACHMENTS

Rod end bearings should incorporate grease fittings if used in high temperature areas.

Provide moisture sealing and good lubrication of all pivot bearings. An adequate number of grease fittings must be provided for forced purging of old grease and foreign materials from heavily loaded bearing areas.

6.2 MATERIAL SELECTION

The selection of materials for structural components will vary depending on environment, load intensity, density, fatigue spectrum, fabrication limitations, electrical properties, and stiffness requirements. All these factors must be considered when selecting the most efficient material to use.

Table III provides material recommendations for primary structure with advantages and limitations considering the factors noted above.

Table III

MATERIALS RECOMMENDATIONS FOR PRIMARY STRUCTURE

ALUMINUM ALLOYS				
Environment	Candidate Materials	Typical Applications	Advantages	Limitations
Over 200° F to 350° F	2024-T6, T62 T81, T851, T8510, T8511, T852	Fuselage skins, empenage skins; frames, stringers, etc., in contact with wing and fuselage skins	1. Relatively small property losses at moderate operating temperatures 2. Resistant to stress-corrosion and exfoliation corrosion 3. Highest strength aluminum alloy in general use at temperatures over 200° F	1. Poorer fracture toughness than 2124 or 2219 2. Limited formability in artificially aged temper 3. Maximum thickness of plate 1.000 in.
	2124-T851	Heavy plate for wing skins, plate stock for hog-outs	1. Provides heavy plate with improved ductility in the short-transverse direction. 2. Fracture toughness improved over 2024	1. Availability restricted to plate over 1.001 in. thickness and to mill order basis only
	2219-T6, T62 T81, T851, T8510, T8511, T852	Wing skins, longerons, structural fittings	1. Best toughness of temperature resistant alloys of aluminum 2. Weldable 3. More resistant to stress-corrosion cracking than 2024 and 2124. 4. Resistant to exfoliation.	1. Lower strength than 2024 or 2124 2. Not readily available in all product forms without special minimum orders
	7000 Series Alloys (General)	General	1. Higher remaining room temperature strengths than 2000 series alloys if elevated temperature exposure	1. Design values must be reduced to compensate for temperature effect. (Resulting values may be

Table III

MATERIALS RECOMMENDATIONS FOR PRIMARY STRUCTURE (CONT)

ALUMINUM ALLOYS (CONTINUED)				
Environment	**Candidate Materials**	**Typical Applications**	**Advantages**	**Limitations**
			is very limited. (Do not consider for general elevated temperature use)	lower than for 2000 series alloy)
-65F thru 200F	7075-T73, T7352, T7351, T3510, T73511	Forgings, fittings, parts receiving heavy machining (such as spars)	1. Maximum resistance to stress-corrosion cracking. 2. Resistant to exfoliation. 3. Good fracture toughness	1. Lower strength than 7075-T6 or T76. 2. Closer in-house processing controls required than for T6.
	7075-T76, T7651, T76510, T76511	General use for interior structure not exposed to elevated temperature. Exterior skins where temperatures are not excessive and advantage can be taken of strength	1. Exfoliation resistant. 2. Stress-corrosion resistance much better than T6. Sufficient for most applications. 3. Strengths above T73 4. Intermediate fracture toughness.	1. Lower strength than T6. 2. Closer in-house processing controls required than T6. 3. T73 preferred for applications most critical for stress corrosion
	7075-T6, T6510, T6511	<u>Material 0.125 inch and under in thickness:</u> All parts fabricable from sheet or extrusions 0.125 inch or less in gauge or thickness should be specified as 7075-T6. Where machining is used to reduce plate or thicker extrusions to below 0.125 inch, the following will apply:	1. Good weight-strength relationship.	1. Susceptible to stress-corrosion cracking and exfoliation corrosion. 2. Not recommended for fracture critical parts.

Table III

MATERIALS RECOMMENDATIONS FOR PRIMARY STRUCTURE (CONT)

ALUMINUM ALLOYS (CONTINUED)

Environment	Candidate Materials	Typical Applications	Advantages	Limitations
		<u>Material in the range 0.125 to 0.250 inch in thickness:</u> 1. <u>Machined:</u> Components machined from sheet or extrusion in this thickness range should be specified as 7075-T76 irrespective of type of application. Removal of surface metal by mechanical or chemical milling to expose underlying metal and grain boundaries increases susceptibility to corrosion as compared to material with the as-fabricated "skin" left essentially intact. This would apply to sheet or thin plate surfaces machined or chemically milled, and to extrusions surface machined or sculptured, but not necessarily to members merely blanked and drilled for fastener holes. The latter are discussed in the next section.		

Table III

MATERIALS RECOMMENDATIONS FOR PRIMARY STRUCTURE (CONT)

ALUMINUM ALLOYS (CONTINUED)

Environment	Candidate Materials	Typical Applications	Advantages	Limitations
		2. <u>As Fabricated:</u> Components produced from sheet or extrusions left essentially in the as-fabricated condition (merely trimmed and drilled) should be specified as 7075-T6, where the application is interior structure, protected from moisture, well drained, and is accessible for inspection and repair. For exterior applications, or internal corrosive environments such as fuel tank sump areas, or other areas subject to collection of moisture, or where internal structure is difficult to inspect or repair, thv details should be specified as 7075-T76. <u>Material over 0.250 inch in thickness:</u> 7075-T6 should not be used in sections 0.250 inch and over in any product form.		

Table III

MATERIALS RECOMMENDATIONS FOR PRIMARY STRUCTURE (CONT)

ALUMINUM ALLOYS (CONTINUED)

Environment	Candidate Materials	Typical Applications	Advantages	Limitations
	7075-T736, T73652, 7175-T736, T73652	Forged fittings and hand forgings only.	1. Strength level comparable to 7075-T6 forgings. 2. Toughness comparable to T73. 3. Stress-corrosion cracking comparable to -T76.	1. Cost 15% to 400% over T6. 2. Limited sources. 3. Producibility in T736 must be verified early in design because of limits on proportions.
	2024-T3, T351, T4, T3510, T3511	Light structural bracketry, fatigue critical webs not exposed to temperatures over 200°F	1. Fair formability in naturally aged condition. 2. High fracture toughness. 3. Good fatigue characteristics.	1. Not recommended for heavy parts susceptible to stress-corrosion cracking. 2. Strengths are typically lower than 7000 series alloys. 3. Inadvertent exposures over 250F could seriously reduce corrosion resistance.
	2024-T851, T852, T8510, T8511	Fittings	1. Better stress-corrosion and exfoliation resistance than T3(X). 2. Can accept temperature overrun where 7075 will have some degradation of properties.	1. 7000 series alloys provide better static strength and better stress-corrosion resistance in T73(X) tempers. 2. Not recommended unless temperature precludes use of 7000 series alloys. 3. Not recommended for fracture critical parts.

Table III

MATERIALS RECOMMENDATIONS FOR PRIMARY STRUCTURE (CONT)

TITANIUM ALLOYS				
Environment	Candidate Materials	Typica' Applicati·ns	Advantages	Limitations
-65F thru +500F	6Al-4V			
	Cond A (sheet, extrusion)	Skins, sine wave beams, stringers, longerons, etc.	1. Preferred for pri· y titanium structure 2. High fracture toughness and fatigue strength 3. High corrosion resistance 4. Weldable	1. Lower strength than Cond STA 2. Subject to dissimilar metals and fretting considerations 3. Higher cost than most steels and aluminum alloys
	Cond RA (plate, bar, billet, forgings)	Machined planks, spars, ribs, fittings, frames, etc.		
-65F thru +500F	6Al-4V Cond STA (all product forms)	Skins, sine wave beams, stringers, longerons, machined planks, spars, ribs, fittings, frames, etc.	1. Higher strength than Cond A 2. Good corrosion resistance 3. Weldable	1. Not recommended for fracture critical applications 2. Improved strength over Cond A limited to 2-inch section thicknesses 3. Higher raw material and processing costs.

Table III

MATERIALS RECOMMENDATIONS FOR PRIMARY STRUCTURE (CONT)

STEEL ALLOYS				
Environment	Candidate Materials	Typical Applications	Advantages	Limitations
-65F to 700F	HP9-4-20 190 ksi min. Plate, bar, forgings	Slat tracks, horiz. stabilizer spindle, fittings	1. High fracture toughness 2. Good machinability 3. Good weldability	1. Must be protected from corrosion 2. Plate availability by mill order only
-65F to 700F	HP9-4-30 220 ksi min. Plate, bar, forgings	Hydraulic actuators, hooks, latches, pins, bearings	1. Good fracture toughness 2. Good machinability 3. Weldable	1. Must be protected from corrosion 2. Plate availability by mill order only
-65F to 500F	300M, 260 ksi or 280 ksi min.	Landing gear cylinders	1. Ultra-high strength 2. Good machinability	1. Must be protected from corrosion 2. Moderate fracture toughness 3. More stringent process controls required during fabrication
-65F to 350F	9315 Carburized (R_C60) Bar, forgings	Gears, splines, ballscrews	1. High wear resistance 2. Good core toughness 3. Good machinability	1. Must be protected from corrosion 2. Core limited to 180-200 ksi tensile strength 3. 2-inch maximum heat treat thickness
-65F to 600F	PH13-8Mo, H1000, H1050, H1100, H1150 Plate, bar, forgings extrusions	Longerons, wing pivot bearings, flap tracks, hooks, struts, fittings, pins, fasteners	1. Corrosion resistant 2. High fracture toughness	1. Plate and extrusion availability only by mill order 2. Slightly higher material cost than other steels

Table III

MATERIALS RECOMMENDATIONS FOR PRIMARY STRUCTURE (CONT)

Environment	Candidate Materials	Typical Applications	Advantages	Limitations
STEEL ALLOYS (CONTINUED)				
			3. Good stress-corrosion resistance 4. Simple, low-distortion heat treatment 5. Weldable	3. Undergoes approx .0005 (H1000) to .0010 (H1100) in./in. shrinkage during heat treatment
-65F to 600F	PH14-8Mo SRH1050 Sheet	Frames, skins (to .125 thick) clamps, brackets stiffeners	1. Corrosion resistant 2. Good fracture toughness 3. Good stress-corrosion resistance 4. Good formability and weldability	1. Availability in sheet gages only through .125 inch thickness 2. Availability by mill order only
-65F to 600F	PH15-7Mo RH1075 Sheet, plate tubing	Brackets, stiffeners, springs, struts, skins (.126 to .25 in. thick)	1. Corrosion resistant 2. Good stress-corrosion resistance 3. Good formability and weldability	1. Limited to .25 inch maximum plate thickness
-65F to 600F	17-4PH H1025, H1150 Bar, forgings	Spacers, bushings, fittings, supports, bellcranks	1. Corrosion resistant 2. Good stress-corrosion resistance 3. Simple, low-distortion heat treatment	1. Not recommended above 155 ksi (H1025) strength level 2. Not recommended for applications over 4 inches in thickness (see 15-5PH)

Table III

MATERIALS RECOMMENDATIONS FOR PRIMARY STRUCTURE (CONT)

STEEL ALLOYS (CONCLUDED)

Environment	Candidate Materials	Typical Applications	Advantages	Limitations
-65F to 600F	15-5PH H1025, H1150 Bar, forgings	Housings, fittings, brackets	1. Same as 17-4PH, items 1, 2, and 3 2. Good structural properties in sections over 4 inches in thickness	1. Not recommended above 155 ksi (H1025) strength level 2. Shrinkage similar to 17-4PH

HEAT RESISTANT ALLOYS

Environment	Candidate Materials	Typical Applications	Advantages	Limitations
-65F to 1200F	Inconel 718 Solution treated and aged (180 ksi) Sheet, plate, bar, forgings	Nacelle frames, skins, stiffeners, engine mounts	1. Corrosion resistant 2. Heat resistant 3. Guaranteed fracture toughness (bar and forgings) 4. Weldable	1. Higher cost than other PH CRES alloys 2. Plate availability by mill order only 3. Machinability only fair

6.2.1 FRACTURE MECHANICS

During inspection visits to USAF, Navy, and airline repair facilities several examples of fatigue failure of structure were observed. Many of the failures were the result of approaching the predicted fatigue life of the aircraft. However, some indicated premature failure resulting from poor selection of material or temper and deficiencies in design and fabrication.

Materials which demonstrated less than expected fatigue life in some cases are 7079-T6 and 7075-T6 aluminum alloy, especially in the heavier forged components, 2024-T851 aluminum alloy in fittings, 2024-T8510 aluminum alloy extrusions, D6 AC alloy steel, and 300 M alloy steel. Fracture toughness in most of these materials has been sacrificed for higher strength tempers. Susceptibility to stress corrosion cracking has also played an important part in low fatigue life of these materials.

Improper attention to detail design to reduce stress concentration points caused by sharp fillet radii, abrupt changes in edge trim and thickness, and location of rivets and bolts in critical stress areas also was indicated. Improper stiffening of thin webs, subjected to high frequency vibration or pressure reversals, have also contributed to failures.

Fabrication deficiencies such as sharp irregularities in machined surfaces, improperly drilled holes, and inadvertent gouges or scratches in critical surfaces have caused premature failures in critical parts. In the case of 300 M steel, grinding burns can seriously affect the fracture and fatigue characteristics of the material due to its extremely high heat treat strength.

6.2.2 CORROSION PROPERTIES

An important factor in the selection of structural materials is their resistance to corrosion. There are several types of corrosion which affect the material's ability to perform its designed function. They can be classified into three groups:

- General corrosion
- Localized corrosion
- Cracking

6.2.2.1 General Corrosion

The most common type is general corrosion which acts uniformly over the surface and gradually thins the material reducing its strength. This type is easiest to detect, control, and predict resulting decrease in product life. It is usually caused by exposure to a damp environment or continuous wetting, or by chemical dissolution by acids, salts, or bases.

6.2.2.2 Localized Corrosion

Localized corrosion includes pitting, crevice corrosion, galvanic corrosion, intergranular corrosion, and selective leaching of alloying elements. (See figure 31, example A.) This type also acts in conjunction with erosion, cavitation in hydraulic units, and fretting of highly loaded bearing surfaces.

6.2.2.3 Cracking

The most difficult type to predict is corrosion which causes cracking of the material. It usually works in conjunction with other forces or elements. Corrosion fatigue is initiated at a crack which starts from a corrosion pit and develops to a wedge-shaped crack from continued stress cycles and corrosion action. Stress corrosion cracking is similar in its development, but is the result of combined action of corrosion and static stresses, either residual or applied. (See figure 31, example B.) Another cracking type of corrosion is exfoliation. This is a subsurface corrosion in the form of cracks that propagate approximately parallel to the surface. It appears most frequently in aluminum alloys, and creates a laminated, flaky, or blistered condition.

6.2.3 FINISHES

Nearly all the materials used in the airframe structure are now given some type of finish coating. The primary purpose of a finish coating is to protect the base material from deterioration. It may also be used to provide surface properties that are suitable for a particular kind of functional service. There are many different types of coating, all of which can be classified into four categories as follows:

- Metallic or metal alloy coatings
- Ceramic and cermet coatings

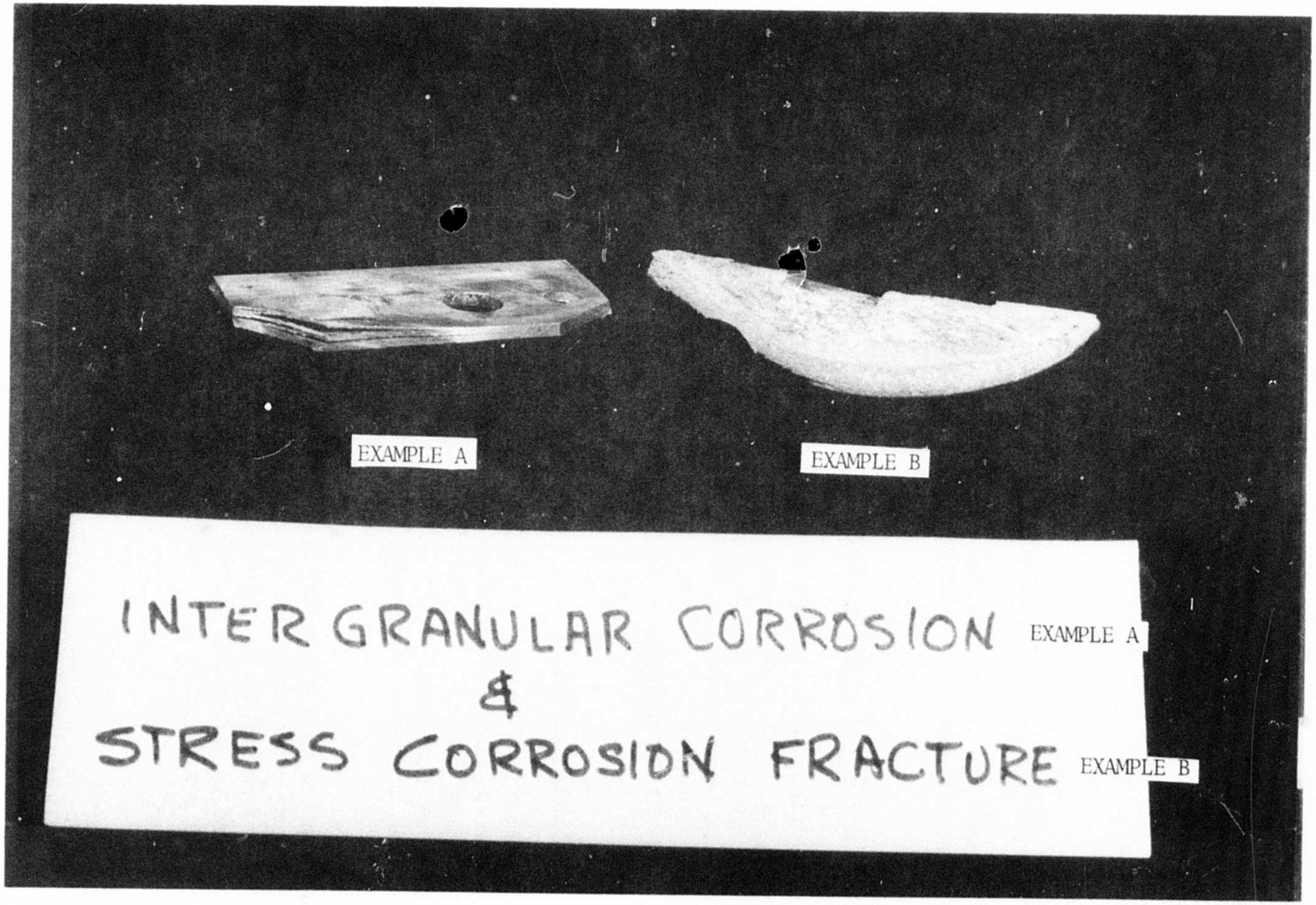


Figure 31. Examples of Corrosion

- Conversion coatings
- Organic coatings

6.2.3.1 Metallic or Metal Alloy Coatings

Metallic and metal alloy coatings are used primarily on alloy steel parts for corrosion resistance or wear. In some cases they are used for electrical conductance.

6.2.3.2 Ceramic or Cermet Coatings

Ceramic and cermet coatings are used on alloy steel parts for resistance to erosion and wear as well as high heat and flame.

6.2.3.3 Conversion Type Coatings

Conversion type coatings are used for corrosion resistance on such base metals as aluminum, iron, carbon steel, magnesium, titanium and zinc. These coatings are formed by converting a thin film of base metal, either by chemical or by electrochemical modification. The surface film is generally composed of either a chromate, oxide, or phosphate compound derived from the base metal and processing solutions.

6.2.3.4 Organic Coatings

The most commonly used finishes are the organic coatings. These comprise all single or multilayer surface films that are formed by solidification of a drying oil or resinous liquid. They are classed as primers, paints, enamels, lacquers, or varnishes. Their use covers several functional purposes such as appearance, corrosion resistance, and reflection or absorption of heat and light.

There are many kinds and compositions of organic coatings, and selection should be made only after consulting various manuals and listings which are available from suppliers. However, for corrosion protection of all aluminum alloys used in structural applications, a good polyurethane epoxy-polyamide coating is preferred by most maintenance personnel.

6.3 CORROSION CONTROL

6.3.1 SELECTION CONSIDERATIONS

The primary consideration in the design and construction of aerospace weapons systems is the ability of the design to comply with structural and operational requirements. In addition, the aerospace weapons are expected to perform reliably and require minimum maintenance over a specified lifetime, which includes minimizing the rate of deterioration. Therefore, in the selection of suitable materials and appropriate processing methods to satisfy structural requirements, consideration must also be given to those materials, processing methods, and protective treatments which reduce service failures due to deterioration of parts and assemblies in service. Deterioration modes which contribute to service failures include, but are not limited to, pitting corrosion, galvanic corrosion, exfoliation corrosion, stress corrosion, corrosion fatigue, thermal embrittlement, fretting fatigue, oxidation, hydrogen embrittlement, weathering and fungus growth. In the entire design phase, attention should be given to precautionary measures to minimize deterioration of individual parts and assemblies as well as the entire system.

Precautionary measures include proper selection of materials, limitation of design operating stresses, relief of residual stress levels, shot peening, heat treatments which reduce corrosion susceptibility, and protective coatings and finishes.

6.3.2 EXCLUSION OF RAIN AND AIRBORNE SPRAY

The design of the system should be such as to prevent water leaking into, or being driven into, any part of the system either on the ground or in flight. All windows, doors, panels, canopies, etc, should be provided with sealing arrangements such that the entry of water is prevented when these items are correctly closed. Particular care should be taken to prevent the wetting of equipment, and heat and sound proofing materials. Sharp corners and recesses should be avoided, so that moisture and solid matter cannot accumulate to initiate localized attack. Sealed floors with suitable drainage should be provided for galleys, toilets, and cockpits.

6.3.3 VENTILATION

Adequate ventilation should be provided in all areas to prevent moisture retention and buildup.

6.3.4 DRAINAGE

Drain holes should be provided in the system to prevent collection or entrapment of water or other unwanted fluids which can enter by various methods. All designs should include considerations for the prevention of water or fluid entrapment, and insure that drain holes are located to effect maximum drainage of accumulated fluids. Actual aircraft configuration and attitude should be considered in addition to component design.

6.3.5 DISSIMILAR METALS

The use of dissimilar metals (as defined by MIL-STD-889) in contact with each other should be limited to applications where similar metals cannot be used due to peculiar design requirements. When it is necessary to use dissimilar metals in contact, the metals should be adequately protected against galvanic corrosion. Galvanic corrosion can be prevented by interposition of a material which will reduce the overall electrochemical potential of the joint or by interposition of an insulating or corrosion inhibiting material.

6.3.6 ALLOY SELECTION

Whenever the design requires the selection of aluminum for structural components, maximum use should be made of alloys, heat treatments, and claddings which minimize susceptibility to pitting, and intergranular or stress corrosion. The following are alloy temper recommendations for resistance to exfoliation or stress corrosion.

<u>Exfoliation resistance</u>

Alloy	Temper
2014, 2024, 2124, 2219	All Artificially Aged
7049	T76XX, T73XX
7050	T76XX, T736XX
7075	T76XX, T736XX

Alloy	Temper
7175	T76XX, T736XX
7475	T76XX, T73XX

Stress corrosion resistance

Alloy	Temper
2024, 2124, 2219	All Artificially Aged
7049	T73XXX
7050	T73XXX
7075	T73XXX
7175	T73XXX
7475	T73XXX

All aluminum sheets used in external environments and interior corrosive environments should be clad on both sides, except where the design requires surface metal removal by machining or chemical milling, or where the design requires adhesvie bonding, or where the design uses alloys of the 5000 or 6000 series type. Surfaces from which cladding has been removed should be protected in accordance with MIL-F-7179, which requires a chemical or anodic film followed by an organic finish.

6.3.7 ALUMINUM ALLOY SELECTION LIMITATIONS

Mill product forms of aluminum alloys 2020, 7079, and 7178 in any temper conditon should not be used for structural applications. The use of 7075-T6 should be limited to thicknesses not to exceed 0.125 inch.

6.3.8 MAXIMUM METAL REMOVAL

Maximum metal removal from surfaces of non-stress relieved structural parts after final heat treatment should not exceed 0.150 inch, unless the

final temper or condition has been demonstrated to have a stress-corrosion resistance of 25 ksi or higher in the short transverse grain direction, as determined by a 20 day alternate immersion test given in FED-STD-151, Method 823. This requirement is applicable to 2000 and 7000 series alloys, but 30 days shall be used on 2000 series alloys. Tension stress-relieved or compression stress-relieved aluminum products should be used wherever possible. Maximum metal removal requirements are not intended to apply to mechanically stress-relieved products because of the low level of internal stresses resulting from mechanical stress-relieving.

6.3.9 SHOT PEENING FOR STRESS CORROSION RESISTANCE

All surfaces of all structural forgings, where accessible after final machining and heat treatment, should be completely shot peened using a minimum of two coverage passes, or placed in compression by other suitable means. This process is not necessary for forgings having a demonstrated stress corrosion resistance of 25 ksi or higher in the short transverse direction, and web areas under 0.080 inch thick where no short-transverse grain is exposed by machining. Those areas of forgings requiring lapped, honed, or polished surface finishes for functional engineering requirements should be shot peened prior to such surface finish operations. Aluminum forgings used in corrosive environments should have essentially no residual surface tensile stresses in the final heat treated and machined condition. Surface finish clean-up of shot peened surfaces such as landing gear bores, as required for proper fit, should not exceed 0.003 inch of surface material removal for aluminum alloys, or 0.0015 inch for steels.

6.3.10 STRESS CORROSION FACTORS

High strength aluminum alloy parts should be designed, manufactured, assembled, and installed so that sustained residual tensile stresses are minimized to prevent premature failures due to stress corrosion cracking. In cases where such stresses cannot be avoided, corrective practices such as use of stress corrosion resistant alloys and tempers, optimum grain-flow orientation, shot peening, or similar surface working processes should be employed.

6.3.11 LOW-ALLOY, HIGH-STRENGTH STEELS

All low alloy, high strength steel parts, including fasteners, require corrosion protective metallic coatings by a process proven to be non-embrittling to the alloy/heat treatment combination. Applicable metallic coatings and finishes are described in subsequent sections of this document.

6.3.12 LIMITATION ON USE OF PROTECTIVE METALLIC COATINGS

Soft surface coatings such as cadmium, nickel-cadmium, and aluminum should not be used for sliding or wear applications. Cadmium plated surfaces should not be used in applications where surface temperature exceeds 450°F. Cadmium should not be used in contact with fuel, hydraulic fluid, or lubricating oil. The use of chrome plating for corrosion protection of alloy steel wear surfaces in interior environments is acceptable. For applications involving exposure to the exterior environment, chrome plating should be considered an acceptable corrosion protection of alloy steel wear surfaces only when the chrome plating is periodically lubricated (fluid or grease types only) or a 0.0015 inch minimum layer of nickel plating is applied under the chrome. All chrome plated steel surfaces should be shot peened prior to plating. Chrome plated surfaces should not be used in applications where service temperatures exceed 700°F.

6.3.13 STRESS CORROSION FACTORS

Alloy steel parts heat treated to 200,000 psi and above should be designed, manufactured, assembled, and installed such that sustained residual surface tensile stresses will be minimized to prevent premature failures due to stress corrosion cracking. Whenever practicable, the use of press or shrink fits, taper pins, clevis joints in which tightening of the bolt imposes a bending load on the female lugs, and straightening or assembly operations that result in sustained residual surface tensile stresses in these materials shall be avoided. In cases where such practices cannot be avoided, apply protective treatment such as stress relief heat treatments, optimum grain-flow orientation, wet installed (with a protective material) inserts and pins, and shot peening or similar surface working techniques to minimize the hazard of stress-corrosion cracking or hydrogen embrittlement damage.

6.3.14 CORROSION-RESISTANT STEELS

Except for the 400 Series Martensitic steels, corrosion resistant steels generally exhibit excellent corrosion resistance and do not require protective coatings for general protection against corrosion. Corrosion resistant steels should be passivated. Table IV may be used as a guide in the selection of corrosion resistant steels for structural applications.

6.3.15 CORROSION-RESISTANT STEEL LIMITATIONS

No corrosion resistant precipitation hardening steels should be used in the H900 condition. Corrosion resistant steels such as maraging, Almar

Table IV

CORROSION CHARACTERISTICS OF CORROSION-RESISTANT STEELS

Class	Alloy	General Corrosion Resistance	Stress Corrosion Resistance
	316	Excellent	Excellent
	347	Excellent	Excellent
	A286	High	Excellent
Austenitic	321	High	Moderate
	304 (ELC)	Moderate to high	Moderate
	302	Moderate	Low
	303	Low to moderate	Low
	440C	Moderate - sensitive to hydrogen embrittlement	All grades susceptible to stress corrosion cracking
Martensitic	420 410 416	Low to moderate - will develop superficial rust film with atmospheric exposure	
	PH13-8Mo	High	Susceptibility varies significantly with composition, heat treatment, and product form
	PH15-7Mo	High	
	PH14-8Mo	High	
Precipitation hardening	17-4PH	High	
	15-5PH	High	
	AM355	High	
	AM330	High	

series, Custom series, etc, should not be heat treated to their highest strength condition. Corrosion resistant steels 19-9DL and 431 should not be used for any application. Series 400 martensitic grade corrosion resistant steels should not be used in the 150,000 to 180,000 psi strength range. Unstabilized austenitic steels may be used up to 700°F. Welded assemblies thereof should not be used unless they have been given a solution heat treatment after welding (except for the stabilized grades 321 and 347, ELC 304 and ELC 316).

6.3.16 SURFACE CONSIDERATIONS

The surfaces of titanium mill products (sheet, plate, bar, forging, and extrusion) should be 100 percent machined or chemically milled to remove all contaminated zones and layers formed while the material was at elevated temperature. This includes contamination as a result of mill processing, heat treating and elevated temperature forming operations.

6.3.17 FRETTING

Titanium alloys are peculiarly susceptible to the reduction of fatigue life due to fretting at interfaces between titanium alloys or titanium and other base metal parts. In any design where fretting is suspected, tests should be made to determine whether such a condition will exist. Design considerations should be applied to minimize fretting in structural applications.

6.3.18 SPECIAL PRECAUTIONS

Titanium parts should not be cadmium plated and should not be used in direct contact with cadmium plated parts or tools. Silver brazing of titanium parts and silver plated fasteners for elevated temperature applications should be avoided. All applications of titanium above 600° F should include consideration of the hot salt cracking phenomenon.

6.3.19 MAGNESIUM

Magnesium alloys should not be used unless they are in areas where low exposure to corrosive environments can be expected, and adequate protection systems can be maintained with ease and high reliability. Specific approval of the procuring activity shall be obtained. Magnesium alloys should not be used in primary flight control systems, for landing gear wheels, or for primary structure. They should not be used in any other areas subject to

abuse, foreign object damage, or to abrasion; or in any location where fluid or moisture entrapment is possible. Only aluminum alloy 5056 rivets should be used for riveting magnesium alloy parts. Magnesium surfaces should not be used for electrical bonding or grounding purposes.

6.3.20 BERYLLIUM

In applications where beryllium is an approved material, consideration should be given to suitable protective coatings to protect parts against corrosion. Tests should be conducted to determine suitability of the protective coating under conditions simulating the expected corrosive environments.

6.3.21 MERCURY

Mercury and many compounds containing mercury can cause accelerated stress cracking of aluminum and titanium alloys. Devices containing mercury should not be used on installed equipment, or during production, where spillage can contact these metals.

6.3.22 ADHESIVELY BONDED ASSEMBLIES

Design of adhesively bonded assemblies should preclude the accumulation and entrapment of water or other contaminants within the structure. Post-assembly edge sealing should be used in addition to design techniques which preclude water entry. Perforated or other core configurations which allow moisture transfer should not be used. All adhesively bonded assemblies should be constructed in accordance with MIL-A-83377. Adhesively bonded assemblies should be designed so that normal handling, and other causes of minor damage will not result in edge or other delamination which could lead to moisture entry.

6.3.23 FOAM PLASTICS

Foam plastics should not be used for metal skin stabilization, or as a sandwich core material in structural components, other than all-plastic sandwich parts, low density filler putties, or hollow glass bead (syntactic) foam. Use of these components should be avoided unless rigorous vibration, sonic fatigue, and all life and environmental exposure tests can amply demonstrate a durable product. All components should be completely sealed to preclude contact of fluids with core.

6.3.24 HYGROSCOPIC MATERIALS

Nonwicking, nonhygroscopic gaskets should be used to prevent moisture intrusion. Felt, leather, cork asbestos, or glycol impregnated gaskets should be avoided. The outer edges of laminated assemblies should be sealed to prevent moisture intrusion.

6.3.25 WATER DISPLACING COMPOUNDS

Water displacing compounds may be used to coat metal surfaces against moisture, fingerprints and corrosion. On plated surfaces of electrical devices (including leads, contacts, and terminal posts), the soft film types of such compounds have been found to be effective protection against corrosion at pores or pinholes in the protective plating, a defect frequently found with standard commercial items. The water displacing compounds shall be in accordance with applicable military specifications. Other corrosion preventive compounds must be approved by the procuring activity.

6.3.26 INSULATING BLANKETS

Where thermal-acoustical insulating blankets are required, they should be either procured with a permanent baked on water repellant binder system, or suitably protected with sealant to prevent any moisture absorbed by the blanket from contacting the metal structure. The blankets should be attached to the aircraft structure by means of an adhesive. The blankets below the floor should be separated from the aircraft structure by standoffs, and attached to the standoffs by an adhesive.

6.3.27 CLEANING

Cleaning of the various types of metallic surfaces, prior to application of the surface treatments and coatings, shall be as specified in MIL-S-5002, using materials and processes which have no damaging effect on the metal, which includes freedom from such things as pits, intergranular attack and significant etching. Appropriate inspection procedures should be established. After cleaning, all parts should be completely free of corrosion products, scale, paint, grease, oil, flux, and other foreign materials (including other metals), and should be given the specific treatment as soon as practicable after cleaning. Particular care shall be exercised in the handling of parts to assure that foreign metals are not inadvertently transferred, as may occur when steel is allowed to come into contact with zinc surfaces.

6.3.28 TITANIUM CONTAMINATION

Care should be taken to ensure that cleaning fluids and other chemicals used on titanium alloys are not detrimental to their performance. Substances which are known to be contaminants and can produce stress corrosion cracking include:

a. Hydrochloric acid

b. Trichlorethylene

c. Carbon tetrachloride

d. All chlorides

e. Chlorinated cutting oils

f. Freons

g. Methyl alcohol

6.3.29 SURFACE DAMAGE

Damage to any previously applied surface treatment or protective finish must be repaired. Damage to surfaces which will become inaccessible because of mating with other parts should be touched up prior to mating. Organic coatings used for repair should be the same as those on the undamaged areas.

6.3.30 INORGANIC FINISHES

6.3.30.1 Detail Requirements

Cleaning, surface treatments, and inorganic finishes for metallic surfaces of aerospace weapons systems parts should be in accordance with MIL-S-5002. Those parts, or surfaces of parts, located in corrosion susceptible areas, or which form exterior surfaces of the system, will require chemical finishing to provide maximum corrosion resistance.

6.3.30.2 Aluminum

All unclad parts made from 7000 series aluminum alloys and located on the exterior surface or in an interior corrosive or abrasive environment, should

be sulfuric acid anodized in accordance with MIL-A-8625, Type II. 2000 series aluminum alloys may be anodized in accordance with MIL-A-8625, Type I or Type II, or chemical film treated in accordance with MIL-C-81706. Shot peening of aluminum alloy parts should be accomplished prior to anodic coating. The detrimental effect of anodic coatings on fatigue life should be considered in design.

6.3.30.3 Adhesive Bonding

Face sheets used for adhesive bonding should not be clad in the bond line. All bond line surfaces should be protected against corrosion by the use of MIL-A-8625, Type I, chromic acid anodizing or FPL etch. The treated surfaces should subsequently be coated with a corrosion inhibiting adhesive primer compatible with the adhesive. Other surface treatments may be used with the approval of the procuring activity. Sandwich construction core should have a corrosion resistant finish in accordance with MIL-C-7438.

6.3.30.4 Cadmium Coatings

Cadmium coatings for all steel parts, including fasteners, should have a minimum thickness of 0.0003 inch and should be subsequently treated with a chromate conversion coating. High strength steels, having an ultimate tensile strength of 200,000 psi and above, should be plated with the titanium-cadmium process in accordance with MIL-STD-1500, the vacuum deposition process in accordance with MIL-C-8837, or a similar non-embrittling process, except as noted in paragraph 6.3.11.

6.3.30.5 Magnesium

Magnesium alloys should be treated in accordance with MIL-M-45202 or MIL-M-46080 prior to painting. Hole drilling after finishes have been applied should not be permitted. Any operation which might remove previously applied finishes should not be permitted.

6.3.31 ORGANIC FINISHES

6.3.31.1 General Requirements

All finishes and coatings should be consistent with the requirements of MIL-F-7179.

6.3.31.2 Detail Requirements

The organic finishes or finish systems used should provide the necessary protection against corrosion for all materials used in areas subjected to corrosive environments. All exterior paints and colors should be consistent with thermal design requirements. Marking and color schemes should be in accordance with MIL-M-25047 and T.O. 1-1-4, or as otherwise specified by the procuring activity. The exterior organic finish system should be MIL-C-83286 aliphatic polyurethane over MIL-P-23377 epoxy polyamide primer. This organic finish system is suitable for temperature requirements to 350°F. Interior primer should conform to MIL-P-23377, except in high temperature areas such as engine bays. Where primers are required in high temperature areas, the selected material should be approved by the procuring activity. Integral fuel tank coatings should meet the requirements of MIL-C-27725. All exterior plastic parts which are subject to rain or solid particle erosion should be protected by coatings conforming to specifications MIL-C-83231 or MIL-C-83445. Justification data, including both laboratory and service experience, should be submitted for approval by the procuring activity whenever materials other than those given above are proposed.

6.3.31.3 Organic Finish Application

The MIL-C-83286 aliphatic polyurethane coating should be applied in two coats to a thickness of 0.0018 to 0.0023 inch, for an overall average total topcoat thickness of 0.0020 inch. The MIL-P-23377 primer should be applied to a thickness of 0.0006 to 0.0009 inch, for an overall average primer thickness of 0.0008 inch. Organic finishes should be applied in accordance with MIL-F-18264.

6.3.31.4 Magnesium Surfaces

Magnesium surfaces should receive pretreatment, two coats of primer, and two top coats prior to assembly. Magnesium components should be installed without undergoing any operation such as hole drilling or fit-up, which would damage this finish. All faying surfaces should be sealed with, and all fasteners should be installed wet with a corrosion inhibiting sealant conforming to MIL-S-81733.

6.3.32 ENVIRONMENTAL SEALING

6.3.32.1 General Requirements

Environmental sealing is utilized to provide protection from corrosion by excluding moisture and other corrodants from joints. It is important that

the areas to be coated with sealant be adequately cleaned before sealant is applied.

6.3.32.2 Detail Requirements

All joints and seams located in exterior or internal corrosive environments, including those in landing gear wells, control surface wells, attachment wells, and structure under fairings should be faying surface sealed with sealant conforming to MIL-S-81733, MIL-C-83982, MIL-S-8802, or MIL-S-83430. The MIL-S-81733 specification covers a sealant which contains a soluble chromate content of 3 to 6 percent for corrosion inhibition. For sealing high temperature areas, MIL-S-38249, firewall sealant, should be used. The use of sealants not covered by a Military Specification should be approved by the procuring activity. Removable panels and access doors should be sealed, either by mechanical seals, or separable faying surface sealant MIL-S-8784.

6.3.32.3 Special Considerations

RTV silicone adhesive sealants are occasionally required for specialized applications in aerospace equipment. Sealants conforming to MIL-A-46106 or MIL-A-46146 should be used for these applications. Caution must be exercised when using MIL-A-46106 material since it may cause corrosion due to liberation of acetic acid during curing. The application precautions given in MIL-A-46106 should be followed.

6.3.33 FASTENER INSTALLATION

6.3.33.1 Detail Requirements

All permanently installed fasteners (fasteners not normally removed for regular access of servicing) should be installed with a corrosion inhibiting sealant conforming to MIL-S-81733, where temperature limitations permit. In high temperature areas, up to 350°F, MIL-P-23377, epoxy primer, or a sealant which is suitable for the thermal environment should be used. Fasteners in integral fuel tanks should be installed with wet sealant as specified in MIL-S-8802 or MIL-S-83430. The use of sealants not covered by a military specification should be approved by the procuring activity.

6.3.33.2 Special Considerations

Quick release fasteners, and removable fasteners penetrating exterior surfaces, should be designed and installed so as to provide a seal to prevent

moisture or fluids from entering. Holes for these fasteners should be primed and allowed to dry prior to installation of the fasteners.

6.3.33.3 Titanium Rivets

Titanium rivets installed in titanium structures may be installed dry, unless sealing is required for liquid tightness.

6.3.33.4 Cadmium-Plated Fasteners

Cadmium-plated fasteners should not be used in applications which would bring them into contact with titanium, and titanium fasteners should not be used in applications which would bring them into contact with cadmium plated components. Cadmium plated fasteners should not be used in contact with graphite composites.

6.3.33.5 Monel- and Copper-Plated Fasteners

Monel fasteners or copper-plated fasteners should not be used in contact with aluminum components.

6.3.33.6 Permanent Fasteners

All permanently torqued fasteners should be lubricated with a mixture of 50 percent (by weight) petrolatum, and 50 percent (by weight) molybdenum disulfide MIL-M-7866.

6.3.33.7 Fasteners in Magnesium

Only 5056 aluminum fasteners should be used to fasten magnesium components.

6.3.33.8 Exfoliation Prevention

The use of aluminum coated fasteners is the preferred method for preventing exfoliation in the countersink area of aluminum skins.

6.4 MAINTENANCE

6.4.1 INTRODUCTION

Aircraft structural maintenance constitutes a significant portion of the total cost of maintaining an airplane in service. A figure of 27.3 percent was determined from cost data reported in USAF maintenance reports on several types of aircraft, which include fighters, trainers, bombers, tankers, and cargo carriers. This fact makes it important to consider the maintainability factor carefully in the design of any new aircraft.

There are four primary design aspects which should be questioned. Does the design provide adequate capability for inspection, access, repair, and replacement?

6.4.2 INSPECTION

There are a number of nondestructive inspection (NDI) techniques available depending on the material being inspected, the structural form, and the type of defect being sought. The most common methods are:

a. Visual

b. Penetrant

c. Eddy current

d. Radiographic

e. Magnetic

f. Ultrasonic

Application of each of these methods and their advantages and disadvantages are shown in figure 32.

Inspection is an important part of the overall maintenance effort. It should not be done indiscriminately since it also can cause additional wear and tear on structure and finishes. Inspection schedules are determined from historic failure experience on similar structure, or from predicted failure data obtained from tests. Its prime purpose is to detect the start of a failure before it becomes either costly to repair, or progresses to catastrophic failure.

Special provisions in the design for inspection are not normally required. However, access for personnel and equipment is necessary. In most cases parts

Type of Method Employed	Application	Advantages	Disadvantages
Visual	Detection of surface defects or structural damage in all materials.	Is simple to use in areas where other methods are impractical. Optical aids further enhance this method.	Reliability of this method depends upon ability and experience of inspector. Accessibility required for direct visibility or borescope.
Penetrant	Detection of surface cracks in all metals, castings, forgings, machined parts and weldments.	Is simple to use. Accurate, fast, easy to interpret.	Defect must be open to surace and accessible to operator. Defect may be covered by smeared metal. Part must be cleaned before and after inspection.
Eddy Current	Detection of surface defects in metallic surfaces, cracks, pits, intergranular corrosion, and heat-treat condition. Conductivity measurement for determining fire-dam aged area.	Is useful when checking attachment holes for cracks not detectable by visual or penetrant. Fast, sensitive, porable.	Sensitive to combination of variations, and unwanted ones must be nulled out. Special probes required for each application.
Radiographic (X-Ray)	Detection of internal flaws and defects such as cracks, corrosion, inclusion, and thickness variations.	Eliminates many disassembly requirements. Has high sensitivity and provides a permanent record on film.	Radiation hazard, trained operators. Film processing equipment required. Crack plane must be nearly parallel to X-ray beam to be detected. Electrical source required. Special equipment require to position X-ray tube and film.
Magnetic Particle	Detection of surface or near-surface defects in ferromagnetic materials of any shape or heat-treat condition.	Is simple in principle, easy, portable. Fast method is positive.	Parts must be cleaned before and demagnetized after inspection. Magnetic flux must be normal to defect plane to yield indications.
Ultrasonic	Detection of surface and subsurface defects, cracks, lack of bond, laminar flaws, and thick ness gaging in most metals by pulse echo	Fast, dependable, easy to operate. Results are immediately known, highly accurate, high sensitivity, and portable.	Trained operator required Electrical source required. Crack plan orientation must be known to select wave mode to be used. Test standards required to establish instrument sensitivity.

Figure 32. Nondestructive Inspection Methods

can be inspected on the aircraft, but those requiring penetrant, magnetic, or eddy current techniques usually have to be removed, and require finish removal and special processing. The ultrasonic method is used frequently on the aircraft without altering the part or finish. The radiographic method is used primarily to detect flaws or cracks which have not propagated to the surface.

6.4.3 ACCESS

One of the most important factors affecting maintenance costs is accessibility. Good access is essential for inspection, and all maintenance actions that result. There are four primary levels of access usually required: inspection access, removal access, personnel access, and service access. Following are some of the characteristics of the doors and openings recommended for these access levels.

6.4.3.1 Inspection Access

- Visual - small openings from 1/2 inch diameter for boroscope equipment to 5 inch diameter for light and viewing from outside the cavity.

- Inspection equipment - may require openings up to personnel size for portable x-ray or ultrasonic equipment.

- Compartment access - minimum of 11 x 17 inch oval-shaped opening required. If area inside the door is restricted, larger sizes may be required. Human engineering studies may be necessary to develop specific access values.

6.4.3.2 Removal Access

- Parts replacement - must have sufficient access for removal without undue disassembly.

- Equipment removal - sizes vary with equipment, but a good rule is to have a minimum of 1 inch clearance all around.

6.4.3.3 Personnel Access

- Crew - refer to AFSC DH 2-1 Design Handbook for Aeronautical System Airframe

- Maintenance personnel - minimum of 11 x 17 inch oval opening required for compartment entry. If area inside the opening is restricted

larger sizes may be required. Human engineering studies should be conducted to determine required dimensions.

6.4.3.4 Servicing Access

- Quick access - for reading gauges, resetting equipment, replenishing expendables, etc. Usually small size doors, commensurate with operations, using spring-loaded hinged doors with quick release latches are desired to minimize maintenance efforts.

- Servicing - sizes will vary with access required. Examples are: avionics equipment, armament, engine, and expendables. Doors require quick acting panel fasteners or latches to minimize servicing time.

6.4.4 REPAIR/MODIFICATION

6.4.4.1 Levels of Repair

Structural repair is accomplished at special facilities equipped according to the level of repair done. In the military there are two levels, base level and depot level, which do minor repair and major repair, respectively.

Minor repair or base level type includes replacements, minor fabrication, adjustments, corrosion control, and component or spot refinishing. Inspection consists of visual type and that which requires only small and portable equipment.

Major repair or depot level type includes all manufacturing processes, refurbishment or replacement of major components, repair of primary structure, complete corrosion cleanup, and aircraft resealing and refinishing. All methods of inspection are employed in determining need for repair or replacement.

6.4.4.2 Design for Repairability

- Make fittings and support castings replaceable.

- Allow enough flange width on stringers, longerons, and frames to permit addition of reinforcements.

- Sking thickness should permit addition of patches and reinforcements.
- Allow sufficient edge distance on fasteners to permit oversize or next larger size to be used.

6.4.5 REPLACEMENT

6.4.5.1 Replaceability

Parts are considered replaceable if they can be removed without destruction of attaching elements, and new parts can be installed by simple trim fitting, if required, and attached with removable type fasteners. The attaching areas of the replacing part may be left blank and drilled at installation. Usually such parts as fittings, supports, wing tips, screwed on access doors, fairings, etc, are made replaceable due to their susceptibility to damage or failures.

6.4.5.2 Interchangeability

Parts are considered interchangeable if they can be removed without destruction of attaching elements, and the replacing part can be installed without further fitting or drilling of attaching holes. A matched pattern of attaching holes must exist between part and mounting structure. Interchangeable items should include radomes, canopies, windshields, windows, landing gear, wheel well doors, gun bay doors, weapon bay doors, wings, movable wing tips, control surfaces, empennage components, and engine access doors. These items are also susceptible to damage but usually require hinged attachments, panel fasteners, or large attaching bolts, and require tooled hole locations.

6.4.5.3 Work Area Consideration

The use of thin skins or thin face sheets on honeycomb structure should be avoided in areas of the fuselage, wing or horizontal stabilizer where maintenance personnel may walk or stand. "No-step" placards are no preventive if the area is convenient for maintenance access to nearby equipment requiring servicing or repair.

Thin skins should not be used on horizontal surfaces below maintenance areas, since they may be cut or dented by dropped tools, doors, or equipment.

6.5 LIFE CYCLE COST IMPACT CONSIDERATIONS

6.5.1 INTRODUCTION

There are many factors that must be considered in the design or repair of military aircraft structures in order to realize the most effective design concepts. The measure of such effectiveness must be established through the determination of their impact on the life-cycle costs (LCC) of the aircraft system. Research has shown that over two billion dollars a year is currently being expended by the Air Force on aircraft maintenance. A significant portion, approximately 25 percent of the maintenance expenditures are for structures. Design studies have also shown that significant savings in maintenance costs may be gained through application of the principles described in this handbook. It is essential, therefore, for aircraft structural designers to be constantly vigilant of the impact that each design factor has upon the cost of ownership for an entire aircraft system.

6.5.2 OBJECTIVE

The objective of this section is to provide the structural designer with information and guidance in the methods that may be used to evaluate the impact of specific design or repair concepts on the life cycle costs of an individual aircraft system. It is structured to permit comparative evaluations of candidate structural design approaches on new aircraft and for candidate repair concepts on existing operational aircraft. The methods have been developed to be compatible with the philosophy and cost estimating procedures contained in Air Force Regulation (AFR) 173-10, "Cost Analysis - USAF Cost and Planning Factors," dated 6 February 1975.

6.5.3 LIFE CYCLE COST (LCC) FACTORS

The LCC factors of a military aircraft consist of three basic phases. These are:

- Research and Development Costs

  The costs associated with the research and development of a military aircraft system are those required to design, test, and evaluate the air vehicle system. For aircraft structures, this would include the costs for conducting research and evaluation testing of new materials, processes, and design concepts as part of a specific aircraft system program. Also included are the costs for fabrication of prototype aircraft and the validation tests conducted to ensure that the design requirements have been achieved.

- Acquisition Costs

  Includes production flyaway costs and initial spares, plus other investment costs such as initial training, AGE and training equipment, AGE and training spares, transportation, facilities, and recurring modifications.

- Operating Costs (Cost of Ownership)

  Includes fuel and lubricants, direct base maintenance personnel (pay and allowances of personnel for inspection, maintenance up through base level, and repair up through base level), replenishment spares, depot maintenance, and base operations support and miscellaneous support (indirect operations costs such as pay and allowances of base operations support personnel, vehicular equipment, material support, rents, utilities, communications, printing and reproduction, medical services, and personnel training costs).

The factors of importance relative to structural maintenance and repair are described in the following procedures for new and existing aircraft systems. The specific values for each factor are totally dependent upon the variables of the individual aircraft in question. These must be derived from the data base developed for the individual applications. In the case of new aircraft, historical data on similar structural design concepts may be one of the only sources of information. Such data must be applied with care to ensure that all assumptions and extrapolations are valid.

6.5.4 NEW AIRCRAFT LCC FACTORS

In the design of a new aircraft, a number of structural concepts and choices of materials may be considered. A comparison of their impacts upon the total life cycle costs of the aircraft system is required to permit an evaluation and selection by the system manager. The major cost factors that are of significance in this process are:

6.5.4.1 Structure Weight

Depending upon the type of aircraft, its mission flight profiles, and the expected number of flight hours for its life expectancy, a significant savings in fuel may be realized for a given reduction in weight.

6.5.4.2 Corrosion Control

The difference in number of maintenance man-hours and material required for corrosion inspection, cleaning, repair, refinishing, and replacement of candidate structure concepts must be evaluated, together with the differences in research, development, and acquisition costs.

6.5.4.3 Structure Accessibility

The maintenance man-hours required to gain access to interior structure and other subsystem components for expected scheduled and unscheduled maintemance must be evaluated. This evaluation should consider not only the time established for normal removal and replacement of the structure (including all access panels), but the probability of damage to those structural elements during base maintenance operations that could require additional labor and materials for repair or replacement

6.5.4.4 Aerospace Ground Equipment

The cost of aircraft structure maintenance equipment must be evaluated for each type of structure considered. This includes all types of special repair tools, machines, inspection equipment, etc.

6.5.4.5 Manufacturing

The cost of special tooling, joining, processing, materials, man-hours, etc., required to produce the aircraft structure must be determined for each candidate design concept.

6.5.4.6 Logistics

The differential in costs for spare parts, materials, and hardware to support the maintenance of each candidate design concept must be determined. This includes the cost of transportation, storage requirements, shelf life, etc, that would have any appreciable affect on the logistics costs.

6.5.4.7 Development Costs

The differential in costs required for the research, development, and testing of candidate structural design concepts may be significant for certain types of materials and construction.

The aggregation of all the significant cost factors may be accomplished by a number of cost estimating procedures, utilized by industry and the Government. As long as all the life cycle cost factors related to structures are included, the identification of the most effective structural concept will be realized.

6.5.5 OPERATIONAL AIRCRAFT LCC FACTORS

For existing operational aircraft, determining the impact of candidate repair or modification concepts on life cycle costs is relatively simple and straightforward. The major variables that are of interest are development, labor, and material costs. The development costs are those associated with the research and testing of new repair or modification design concepts, preparation of engineering drawing, time compliance technical orders (TCTO's), kit-proofing, test programs, tooling, etc. These are generally categorized as "nonrecurring" costs. The costs associated with the production of materials for modification or repair "kits" or packages together with the labor required for their installation are considered as "recurring" costs.

An evaluation of the differences in costs between the costs associated with the maintenance or repair of an existing structural condition and a new concept must be conducted to determine the impact upon the life cycle cost of the total aircraft system. This must take into account the estimated remaining life of the aircraft structure and the frequency of the maintenance efforts.

Several examples of typical repair concepts are presented to illustrate the cost factors involved in an evaluation. They are calculated to determine the savings that would be achieved for a remaining life of 10 years and a 3-year "break-even" period in which a repair would pay for itself.

Table V shows a cost breakdown comparison of a low cost design improvement with the existing design for a fuselage urinal area in a large aircraft. Severe corrosion of the substructure was being continually experienced due to leakage and spillage of the urinal unit during operational and maintenance operations. Considerable man-hours were being expended during periodic depot maintenance (PDM) scheduled at 48-month intervals. The labor costs amounted to an average of $10,920 and material costs of $150. The low cost repair consisted of a new shield insert that would contain any spillage and prevent its migration to the substructure. The material cost for the shield is $195. A total of 5 labor hours would be required for installation of the shield at a cost of $100. These two costs represent the recurring costs for each aircraft. The nonrecurring costs consist of the engineering for design of the shield, revisions to existing maintenance handbooks and illustrated parts breakdown documents, preparation of the TCTO, packaging and shipping of kits to the responsible Air Logistics Center, and test and kit-proofing efforts.

Table V

COST BREAKDOWN - PRESENT AND REPAIR DESIGNS - (FORWARD FUSELAGE URINAL AREA)

Effective on 272 aircraft. Programmed depot maintenance (PDM) schedule 48 months.

Present Design				Repair Design			
Part or Matl	Qty	Cost	Total	Part or Matl	Qty	Cost	Total
Misc corrosion control parts & material			\$150.00[1]	Shield	1	\$195.00	\$195.00
Total			150.00	Total			195.00
Instl (labor at \$20/hr	546 hr[2]	\$20/hr	10,920.00	Instl (labor at \$20/hr)	5 hr[3]	\$20/hr	100.00
				Implementation (nonrecurring):			
				Engineering	22,195		
				Hndbk rev	500		
				T.C.T.O prep	1,200		
				Pkg & shipping	21,166		
				Test/kit proofing	520		
				Total nonrec	\$45,581[3]		
Aircraft total cost			\$11,070.00	Aircraft total cost			\$295.00

NOTE:
1. Cost estimate by contractor based on observation at Oklahoma City ALC.
2. Ref estimate by Oklahoma City ALC for corrosion on 2-year fly-in program.
3. Contractor estimate.

Table VI lists the life cycle cost comparisons for a 3-year break-even period and a 10-year remaining life period. As can be seen, a significant savings could be realized for what, at first examination, may have appeared to be relatively simple and unimportant but which can be very productive.

Table VII and VIII illustrate a similar savings for a simple modification to seal a wing joint rib installation to prevent the entrance of water and anti-icing fluids into an aluminum fitting where severe corrosion could occur and require extensive corrosion control effort and, in some cases, removal and replacement of the wing rib. In this case, a life cycle cost saving of approximately $3 million would be realized.

Table IX and X illustrate an example where a jet engine tailpipe clamp would fail frequently, causing potential heat damage to primary structure requiring significant inspection costs and aircraft down time. A relatively inexpensive change to the clamp is shown to reduce the cost of ownership of the aircraft systems by approximately $634,000.

For existing aircraft systems, Air Force Regulation 173-10, "USAF Cost and Planning Factors," should be used to obtain the proper values for specific aircraft depot and base level maintenance costs related to flying hours. These include such factors as:

- Fuel
- Depot maintenance
- Base maintenance materials
- Labor rates
- Replenishment spares

Table VI

LIFE CYCLE COST COMPARISON
(FORWARD FUSELAGE URINAL AREA)

Effective on 272 aircraft.

Cost Description	3 Yr Period Present Cost	3 Yr Period Repair Cost	10 Yr Period Present Cost	10 Yr Period Repair Cost
Nonrecurring:				
Implementation		$ 45,581		$ 45,581
Fleet modification				
Parts/material		53,040		53,040
Labor at $20/hr		27,200		27,200
Recurring:				
Depot maint labor	$4,455,360	371.280	$14,851,200	$ 2,970,240[1]
Base maint labor				
Material	61,200	5,100	204,000	40,800[1]
Spares	0	0	0	0
Spares pkg & shipping	0	0	0	0
Total	$4,516,560	$ 502,201	$15,055,200	$ 3,136,861
Savings:				
Present costs less repair costs		$4,014,359		$11,918,339

1. Based on repair design reducing corrosion control effort and material requirement by 75%.

Table VII

COST BREAKDOWN - PRESENT AND REPAIR DESIGNS - (INNER-TO-OUTER-WING JOINT RIB)

Effective on 759 aircraft. Programmed depot maintenance (PDM) schedule 48 months.

Present Design				Repair Design			
Part or Matl	Qty	Cost	Total	Part or Matl	Qty	Cost	Total
Rib, inner Rib, outer	2 2	$4,233.00 3,827.00	$ 8,466.00[1] 7,654.00[1]	Mylar cover Adhesive	2 1 pint	$ 0.20 1.60	$ 0.40 1.60
Total			16,120.00[1]	Total			2.00
Instl (labor at $20/hr)	2,400 hr[1,3] 36 hr[2]		48,000.00 720.00	Instl (labor at $20/hr)	3 hr[4]	$20/hr	60.00
				Implementation (nonrecurring): Engineering Hndbk rev T.C.T.O. prep Pkg & shipping Test/kit proofing Total nonrec		Total: 2,241.00 400.00 1,000.00 7,348.00 520.00 $11,509.00[4]	
Aircraft total cost			$64,120.00[1] 720.00[2]	Aircraft total cost			$62.00

NOTE: 1. Only for aircraft that require rib replacement. Estimated at three aircraft per year by OCALC.
2. For corrosion control on aircraft that do not require rib replacement. Estimate by contractor.
3. Man-hours provided by OCALC.
4. Contractor estimate.

Table VIII

LIFE CYCLE COST COMPARISON

(INNER-TO-OUTER-WING JOINT RIB)

Effective on 759 Aircraft.

Cost Description	3 Yr Period		10 Yr Period	
	Present Cost	Repair Cost	Present Cost	Repair Cost
Nonrecurring:				
Implementation		$11,509		$11,509
Fleet modification				
Parts/material[3]		1,518		1,518
Labor at $20/hr[3]		45,540		45,540
Recurring:				
Depot maint labor	$835,380[3]		$2,784,600[2]	$136,620[4]
Base maint labor				
Material				
Spares	145,080[1]	152	483,600[1]	152
Spares kg & shipping	4,980[1]		16,600[1]	
Total	$985,440	$58,719	$3,284,800	$195,339
Savings: Present costs less repair costs		$926,721		$3,089,461

1. Based on three aircraft per year requiring rib replacement per OCALC.
2. Includes rib replacement at three aircraft per year and corrosion control on remainder.
3. Based on all 759 aircraft having change accomplished in 1 year.
4. Assumes a residual of 10% of original corrosion costs incurred.

Table IX

COST BREAKDOWN - PRESENT AND REPAIR DESIGNS - (ENGINE TAILPIPE CLAMP)

Effective on 903 aircraft. Clamp replacement schedule every 16 months.

Present Design				Repair Design			
Part or Matl	Qty	Cost	Total	Part or Matl	Qty	Cost	Total
Clamp assy	2	$25.20	$50.40	Spring Bolt & nut Washer	4 4 8	$0.75 4.84 0.14	$ 3.00 19.36 1.12
Total			50.40	Total			23.48
Instl (labor at $20/hr)	2 hr	$20/hr	40.00	Instl (labor at $20/hr	2 hr[1]	$20/hr	40.00
				Implementation (nonrecurring): Engineering Hndbk rev T.C.T.O. prep Pkg & shipping Test/kit proofing		 $ 3,488 300 1,000 13,527 550	
				Total nonrec		$18,865[1]	
Aircraft total cost			$90.40	Aircraft total cost			$63.48

NOTE: 1. Contractor estimate.

Table X

LIFE CYCLE COST COMPARISON
(ENGINE TAILPIPE CLAMP)

Effective on 903 aircraft.

Cost Description	3 Yr Period		10 Yr Period	
	Present Cost	Repair Cost	Present Cost	Repair Cost
Nonrecurring[1]:				
Implementation		$18,865		$18,865
Fleet modification				
Parts/material		21,202		21,202
Labor at $20/hr		36,120		36,120
Recurring:				
Depot maint labor				
Base maint labor	$81,270		$270,900	3,612[2]
Material				
Spares	102,400	2,120	341,334	2,120
Spares pkg & shipping		1,353		1,353
Total	$220,891	$96,203	$736,306	$101,469
Savings: Present costs less repair cost		$124,688		$634,837

1. Assumes repair design installed within 3 years.

2. Spares installation.

6.6 TRADE FACTORS

6.6.1 INTRODUCTION

In the evaluation of two or more candidate methods or solutions there are several trade factors to be considered. These are the effects on:

1. Procurement costs
2. Maintenance costs
3. Performance
4. Strength and rigidity
5. Fatigue life
6. Weight
7. Reliability
8. Safety
9. Logistics
10. Ground equipment and facilities

6.6.2 COMPONENTS OF TRADE FACTORS

- Procurement costs involve costs of design, fabrication, handling, shipping, and installation.
- Maintenance costs include inspection, servicing, corrosion control, rework, and parts replacement due to wear, breakage, or damage.
- Performance of the system is affected primarily by weight deltas; however, it is sometimes affected also by strength and rigidity.
- Strength and rigidity depend on material properties, sizing, and thermal environment.
- Fatigue life is determined by the design minimizing stress concentrations, by the spectrum of stress levels and cycles, and by the fracture mechanics of the material.

- Weight deltas result from the efficiency of the design, strength-weight ratios of materials used, and environmental penalties involved.

- Reliability can be affected by the number of elements in the design and the reliability of each.

- Safety depends on the margin of safety designed into each element, redundancy of load paths, and failsafe design in event of failure.

- Logistics effects include availability, storability, shipping and handling, and number of parts involved.

- Ground equipment and facilities trade factor includes cost and availability of ground equipment, and facility requirement for operation service, maintenance, and repair.

6.7 DESIGN CHECKLIST

The following is a checklist for airframe structural design directed toward alleviating the high cost of maintenance and repair.

6.7.1 PRIMARY STRUCTURE

- Provide for easy removal of major components such as wings, empennage, and fuselage sections.

- Select materials which are resistant to stress corrosion cracking.

- Make highly stressed fittings, frames, or wing spar members replaceable without undue disassembly of structure.

- Do not use magnesium unless substantial weight or other advantages are possible.

- Use commonly available materials such as 2200, 2400, or 7000 series aluminum alloys, 321 or 17-4 CRES steel, and 4130 or 4340 alloy steel.

- Provide bolted manufacturing break joints in large components, such as wings or fuselages.

- Make structural sections interchangeable with corresponding sections of other units where possible.

- Provide access through mold line surface, and frames or ribs, into closed compartments.

- Limit sandwich panel sizes to 4 x 10 feet for compatibility with repair facilities.

- Use generous fillet radii and inside corner trim radii on highly stressed fittings, spars, and skins.

- Avoid sharp steps in skin thickness which are transverse to loading.

- Use tapered shank and interference fit fasteners only where significant advantages in fatigue life are obtained.

- Avoid use of castings for primary structure.

6.7.2 SECONDARY STRUCTURE

- Use commonly available materials for ease of replacement and repair.

- Use magnesium alloy only where it can be easily inspected and replaced.

- Provide for removal of system ducts and shafts for repair or replacement.

- Make parts from sheet metal, where possible, for repairability.

- Provide adequate skin thickness and reinforcement to withstand gun blast effects and acoustic vibration.

- Provide complete drainage paths throughout all unsealed compartments.

- Eliminate or cover all pockets and corners in landing gear wells, where moisture or mud can be trapped, so as to prevent corrosion.

- Seal enclosed areas in wheel well doors to prevent moisture entrapment and corrosion.

- Insulate all joints between dissimilar metals with a nonabsorptive material to prevent corrosion.

- Provide a moisture seal over all floor structure in areas subject to liquid spillage or moisture entrapment by rugs.

(NOTE: Flush floor surfaces are recommended to prevent holes being worn in the moisture seal.)

6.7.3 FASTENERS

- Refer to section 5 for fastener selection.
- Use solid aluminum alloy driven rivets where possible.
- Avoid the use of "ice-box" type rivets.
- Use monel rivets for high temperature applications in titanium or CRES steel. Type A286 rivets may be used if temperature exceeds 800° F. For highest strength and lightest weight the 6-4 Ti bimetal rivet can be used up to 500° F.
- Where blind rivets are used requiring special installation tools, the same type should be used throughout the airplane to simplify repair facilities tool requirements.
- Special high strength permanently installed type fasteners should be limited to a minimum number of types to reduce facilities requirements.
- Use bulb-type stems on pull rivets in inlet ducts to prevent foreign object damage.
- Use threaded fasteners where infrequent removability is required.
- Use cadmium plated alloy steel fasteners in normal applications in aluminum or steel where temperatures do not exceed 450° F. Do not use in titanium.
- Titanium bolts should be used only where weight saving warrants the additional cost.
- Type A-286 CRES steel bolts should be used only in elevated temperature applications above 450° F.
- Head recess on flush bolts should all be of the same type, preferably Hi-torque.

- All permanently installed fasteners on exterior surfaces should be installed with wet primer to prevent corrosion.

- Frequently removed access panels should be attached with quick-acting panel fasteners.

- Panel fasteners should be standardized on the same diameter and head recess to simplify maintenance. Suggest 3/8-inch diameter and hex recess.

6.7.4 CORROSION CONTROL

- Exclude rain and airborne spray.
- Provide adequate ventilation.
- Provide adequate drainage.
- Avoid contact of dissimilar metals as defined by MIL-STD-889.
- Select best aluminum alloy for exfoliation resistance.
- Select best aluminum alloy for good stress corrosion resistance.
- Do not use aluminum alloys 2020, 7079, and 7178.
- Do not exceed 0.150 inch maximum metal removal after final heat treat unless stress relieved.
- Shot peen surfaces of forgings for stress corrosion resistance.
- Apply corrosion protective coating on low alloy, high strength steel parts, including fasteners.
- Corrosion resistant steels must be passivated.
- Select corrosion resistant steels according to general and stress corrosion resistance.
- Remove all mill-processed surfaces of titanium.
- Minimize fretting in titanium structural applications.
- Do not cadmium-plate titanium parts.

- Avoid use of magnesium alloys except in specially approved areas.
- Completely seal adhesively bonded assemblies to preclude water entry.
- Use foam plastics only in all-plastic sandwich parts, and not as metal skin stabilization material.
- Use nonwicking, nonhygroscopic gaskets to prevent moisture intrusion.
- Use water displacing compounds to prevent corrosion.
- Use insulating blankets with permanent baked on water repellent binder system, or protected with sealant to prevent moisture absorption.
- Clean metallic surfaces in accordance with MIL-S-5002 prior to coating application.
- Apply inorganic finishes for metallic surfaces in accordance with MIL-S-5002.
- Use all organic finishes in accordance with MIL-F-7179.
- Apply organic finishes in accordance with MIL-C-83286 and MIL-F-18264.
- For environmental sealing use sealants conforming to MIL-S-81733, MIL-C-83982, MIL-S-8802, MIL-S-83430, MIL-S-38249, MIL-S-8784, MIL-A-46106, or MIL-A-46146.
- Install permanently installed fasteners with corrosion inhibiting sealant conforming to MIL-S-81733.
- Design and install quick release fasteners and removable fasteners so as to provide a seal to prevent moisture or fluids from entering cavity.
- Do not use cadmium-plated fasteners in titanium.
- Do not use titanium fasteners in cadmium-plated components.
- Use only 5056 aluminum fasteners in magnesium components.

7.0 REPAIR OR MODIFICATION DESIGN

7.1 REPAIR OR MODIFICATION DRAWINGS

These drawings must be as simple and clear as possible. Three things must be remembered when making these drawings: (1) the mechanic is working on a completely assembled aircraft, (2) the facilities and equipment available to the mechanic, and (3) the skill level of the mechanic.

Perspective, or isometric-exploded view drawings should be used whenever possible, as they are easier for the mechanic to understand than a plan view drawing and its associated myriad of dotted lines. See figure 33 for an example of plan view drawing of a repair, and figure 34 for the recommended perspective view drawing of the same repair.

All dimensions on a repair or modification drawing must be given from a physical location on the aircraft. Do <u>not</u> use station lines, butt lines, or water lines.

7.2 REPAIR OR MODIFICATION KITS

Kits should be 100 percent complete, including all necessary hardware. Do not call out any items to be Government-furnished, unless directed to do so by the procuring activity. Special fasteners should not be used in these kits, unless the design precludes the use of standard fasteners. DD rivets should <u>not</u> be used in any kit.

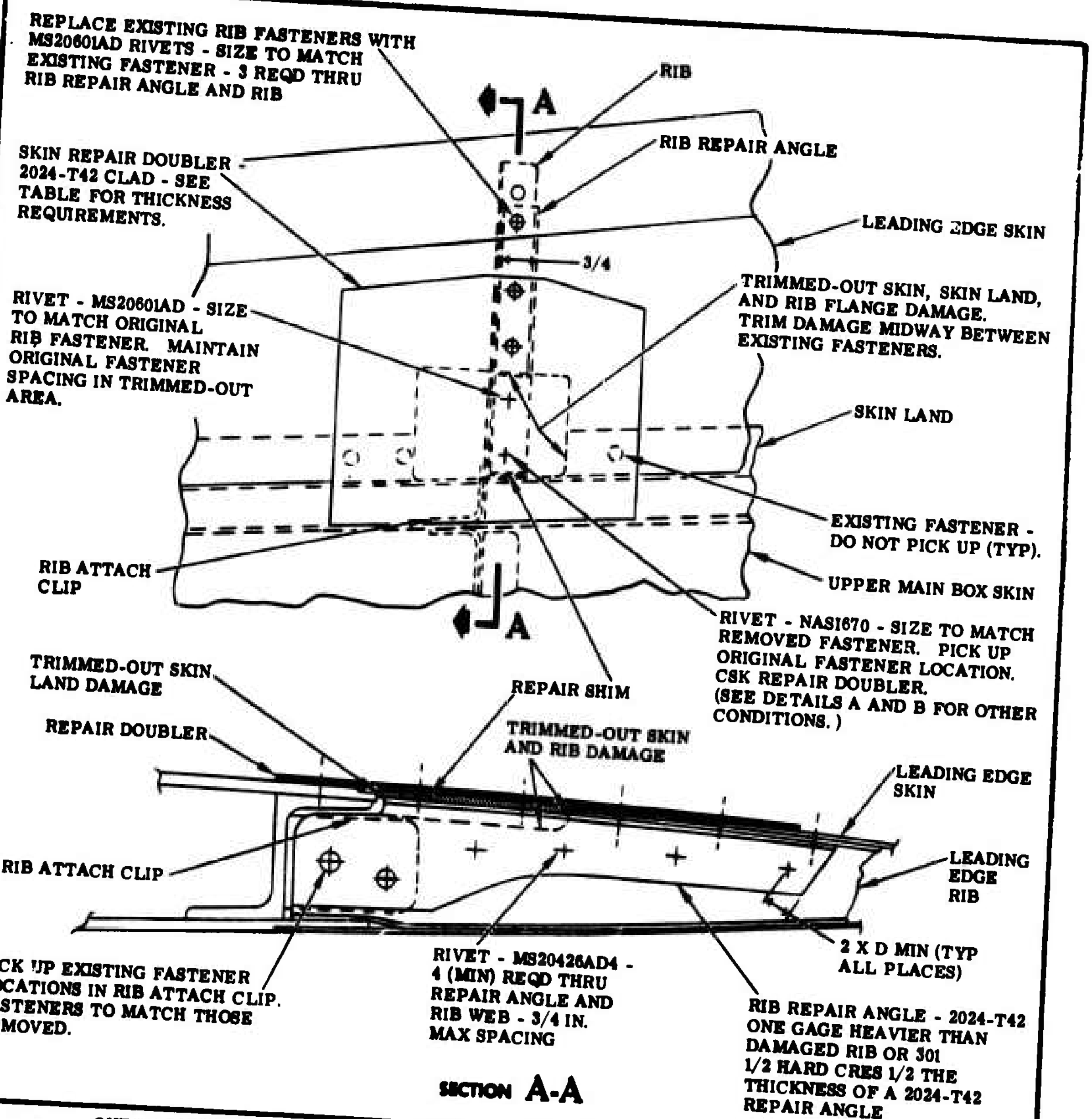
7.3 REPAIR CONCEPTS

7.3.1 MAIN LANDING, GEAR TRUNNION, AND PIN REPAIR CONCEPT (Refer to paragraph 4.1 for problem.)

Cover the openings on upper trunnion surface with a metal cap which is riveted to the trunnion.

Provide additional passageways in the pin to allow grease from the existing lubricating system to be applied to the pin/trunnion bearing surfaces in order to purge out grit which can possibly enter from underneath the trunnion during taxi on wet runways.

Provide a dry lube surface on the trunnion bearing area. Use either cadmium plate or dry lube.



OVER-RIB REPAIR DOUBLER THICKNESS			
SURFACE	LEADING EDGE	TRAILING EDGE	
		STA 95 AND INBD	OUTBD OF STA 95
HOR. STAB.	0.063	0.071	0.063
VERT. STAB.	0.063		

NOTE

See details and notes on sheet 3.

LEGEND

D - Fastener diameter

F-100D-3X-20-7

Figure 33. Horizontal and Vertical Stabilizer Leading Edge and Trailing Edge Skin Repairs

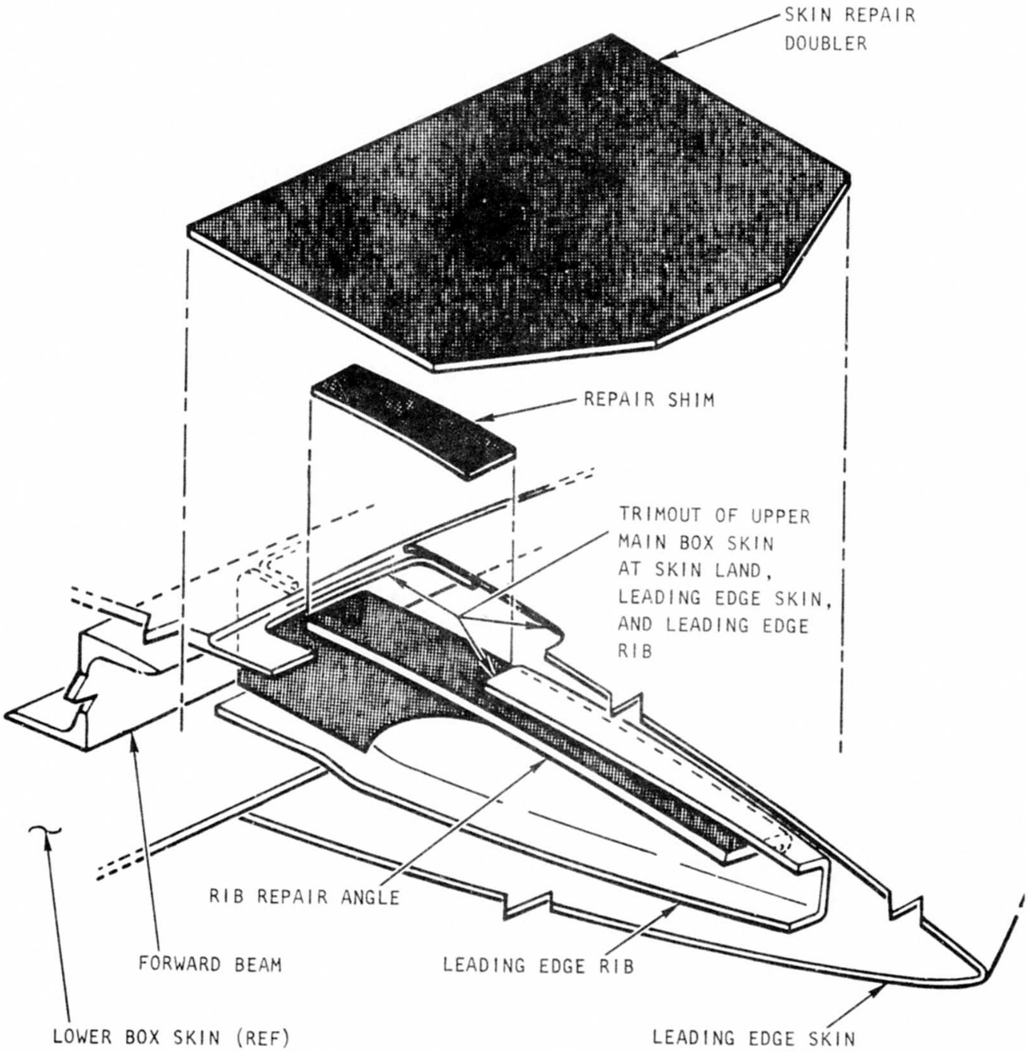


Figure 34. Horizontal and Vertical Stabilizer Leading Edge and Trailing Edge Skin Repairs

If bearing stresses are too high, they can be reduced by adding approximately 20 percent to the bearing width on the aft side of the trunnion, and by reshaping the inner end of the pin to add more bearing surface on the upper and lower sides of the trunnion hole (see figure 35). Existing trunnion fittings which have galled, or damaged bearing surfaces may be salvaged by reaming out the damaged area, and plating or flame spraying to build up the area sufficiently for rehoning the hole to proper diameter.

7.3.2 INNER TO OUTER WING JOINT RIB REPAIR CONCEPT (Refer to paragraph 4.2 for problem.)

Revise the fairing cover land and attachment for a better seal. A secondary seal (a mylar type diaphram), should be cemented to the fittings covering the bath-tub recesses so that moisture is not allowed into the cavities. A desiccant could be also added to the upper cavities to absorb any condensation within them (see figure 36).

7.3.3 GALLEY LAVATORIES AND URINAL AREA REPAIR CONCEPTS (Refer to paragraph 4.3 for problem.)

Cover the wall and floor areas around and below the galley, lavatories, or urinal with a protective shield and floor pan. These items should be fabricated from fiberglass or a similar material. If possible the wall shield and floor pan should be fabricated in one piece. See figure 37 for a recommended protective one piece wall shield and floor pan for a typical urinal installation in a bomber.

7.3.4 ENGINE TAIL PIPE CLAMP (Refer to paragraph 4.4 for problem.)

It is estimated that temperature differential between the tail pipe and the clamp can be as high as 500° F, which requires a spring with 0.11-inch deflection in addition to that which 25-inch-pound torque will provide. It is important to use a spring to keep tension on the clamp and intimate contact with the engine and tailpipe in order to have a high thermal conductivity, so as to reduce the differential as much as possible.

A rectangular section type compression spring, such as is used on tooling dies, can be utilized. Dimensions of approximately 7/8-inch maximum OD, with 11/32-inch ID, and approximately 1-1/8-inch height are preferable (see figure 38).

The tension yield of type 321 CRES is 30,000 psi, which will be reached with only slightly over 100° F differential temperature. The tension load at this point will be approximately 3,000 pounds in the clamp. The bolt and

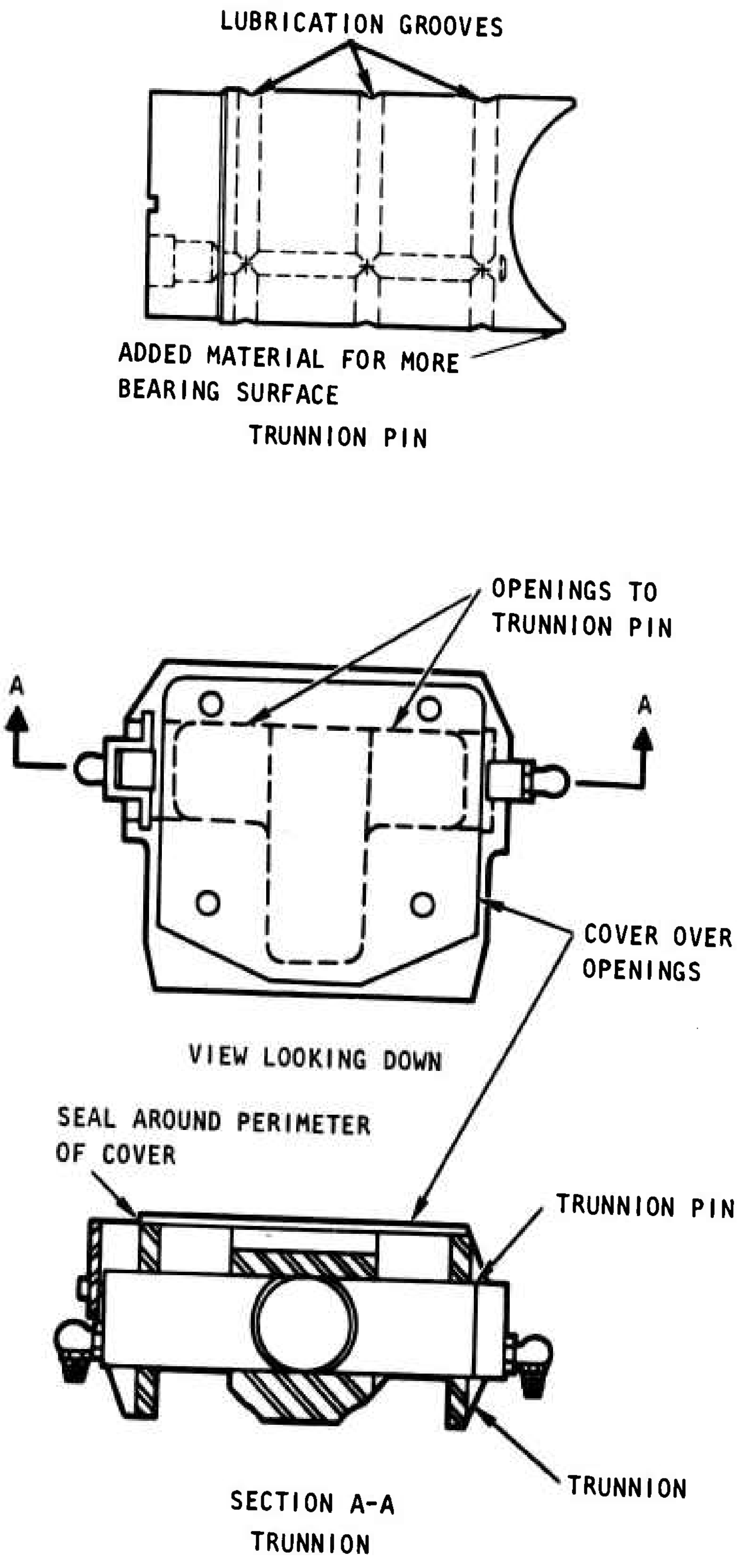


Figure 35. Main Landing Gear Trunnion and Pin Repair Concept

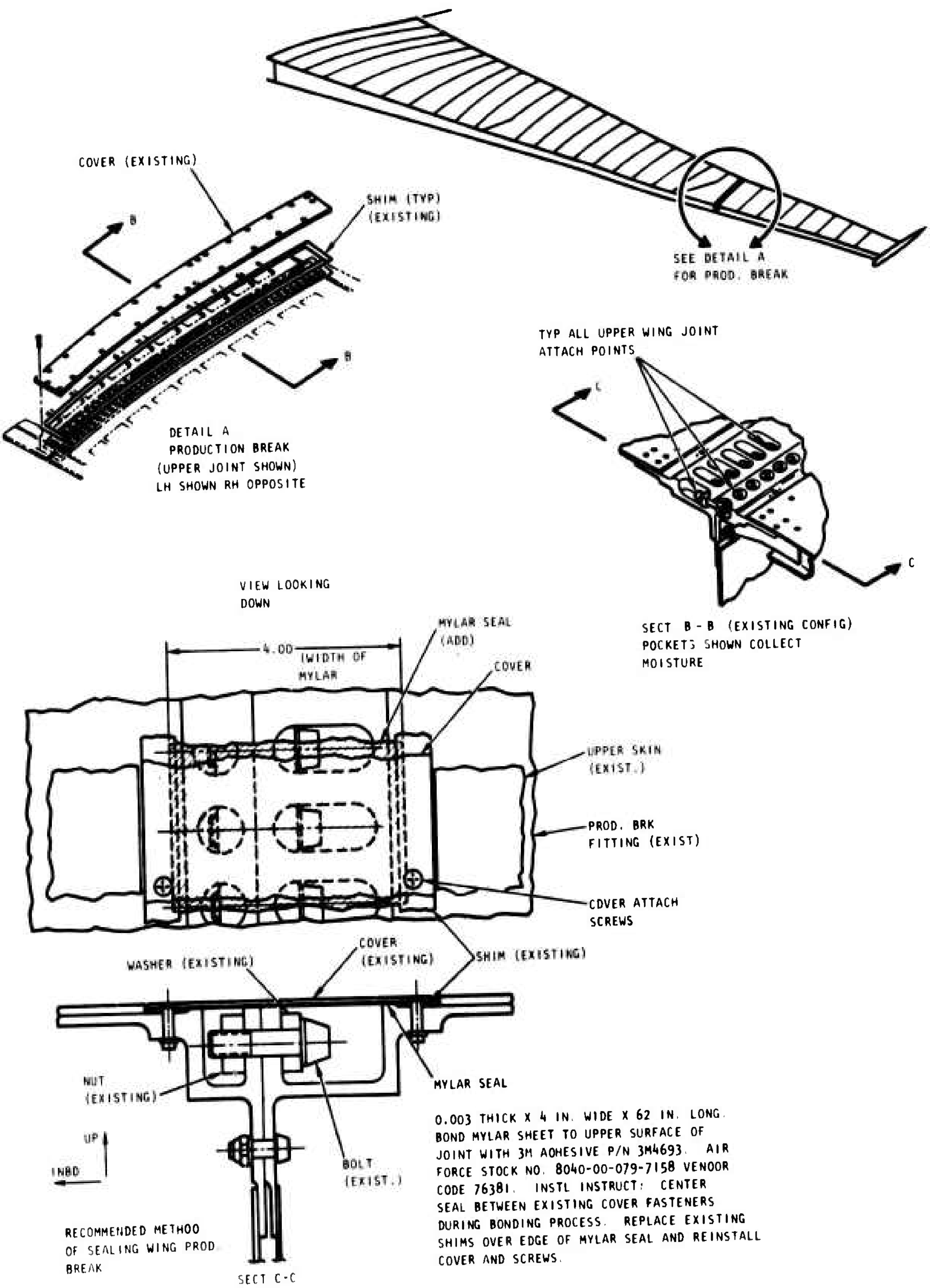


Figure 36. Inner-to-Outer Wing Joint Rib Repair Concept

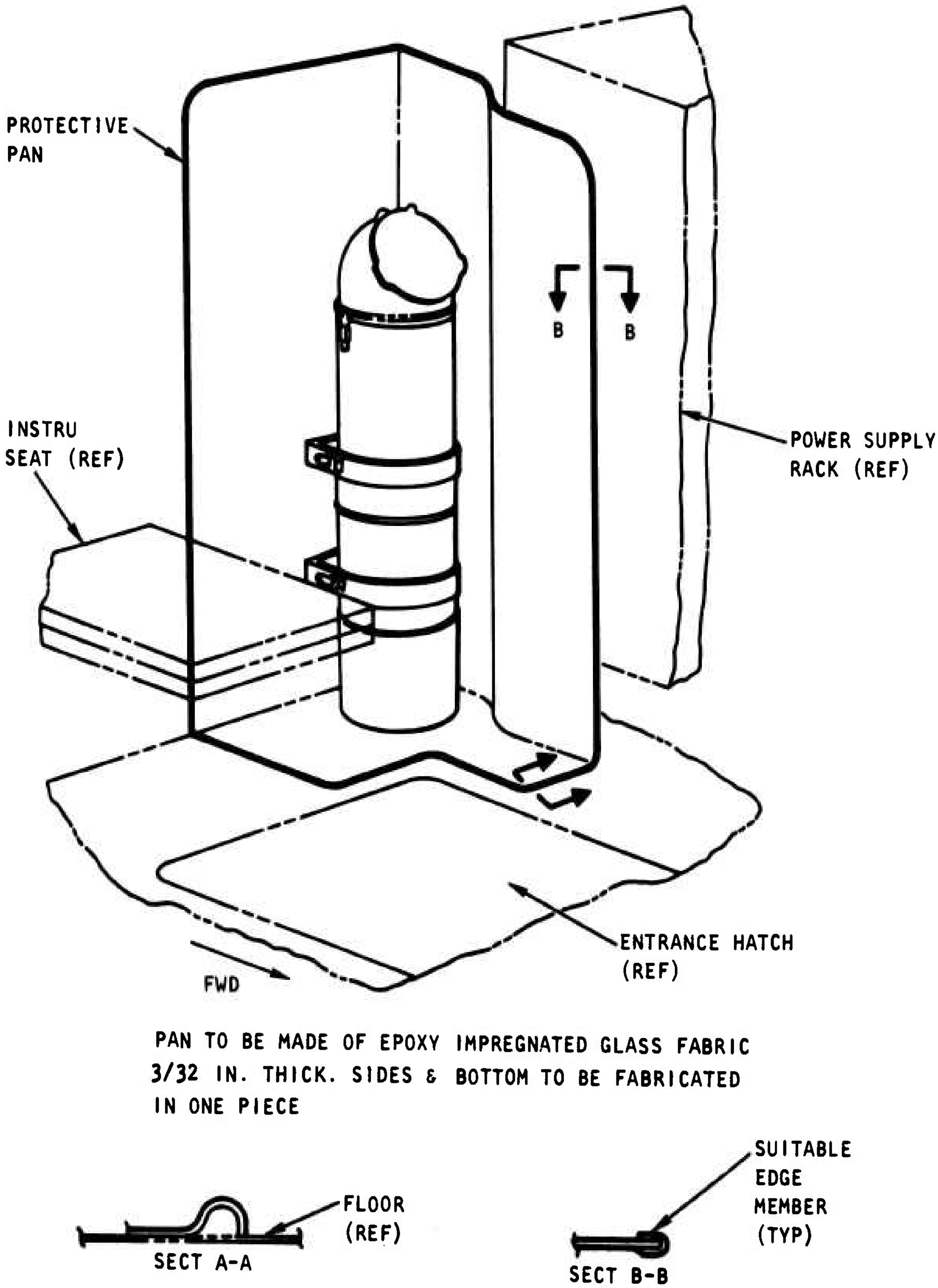


Figure 37. Typical Bomber Urinal Area Repair Concept

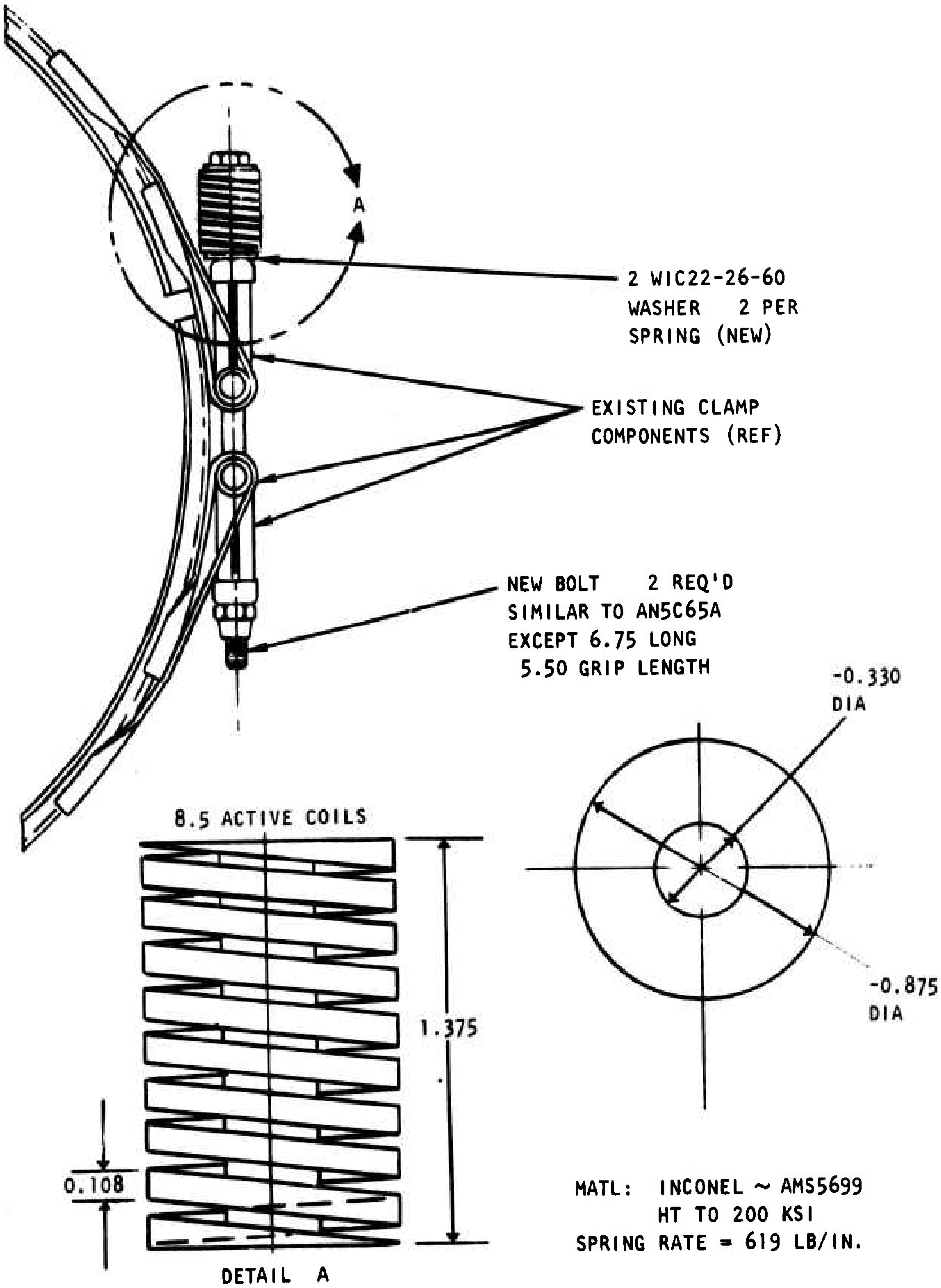


Figure 38. Rectangular Section Spring Clamp Repair Concept

trunnion are at an angle of 19 degrees to the clamp lug. This will result in 2,690 pounds in the bolt. The spring should be designed for this load when fully compressed.

7.3.5 ENGINE COWL DOOR HINGE FITTING (Refer to paragraph 4.5 for problem.)

Two approaches to the cracked fitting have been considered. The first was a new fitting, with an adequate beefup of the cracked flange area. The increased thickness of the flanges and the addition of gussets will insure adequate load paths for any momentary high stress level the fitting may experience (see figure 39).

An alternate approach to reinforcement of the crack prone area would be by welding. The flanges on either side of the critical area could be built up with weld until the cross-sectional area is sufficient to withstand any stress loads the hinge may feel (see figure 40).

7.3.6 WING SLAT ACTUATOR DOOR REPAIR CONCEPT (Refer to paragraph 4.6 for problem.)

It was proposed to add self-aligning ramp blocks to each side of the doors at the forward and aft ends to prevent a slightly misaligned door from catching on the structure, and to bring it into alignment with the jamb for proper seating during retraction (see figure 41).

It was also recommended that a review be made of the kinematics of the mechanism to assure that the adjusting fittings are of proper length for adequate thread engagement.

7.3.7 NACELLE AFT COWL DOOR REPAIR CONCEPT (Refer to paragraph 4.7 for problem.)

Access to that area of the duct unreachable from either end is mandatory for repair of the vanes and their attachments. Provisions for two small removable doors in the inboard duct wall structure can be made. These doors should be located between the two main vanes, and between frames six and nine for the forward door, and between frames 13 and 16 for the aft door. The two intermediate frames, in each case, require removable splice joints at the upper end and lower edges of each door. Sections of each vane, in the proximity of each door, must also be made removable for access into the upper and lower sections of the duct (see figure 42).

A mechanical attachment of the outboard vane attach member, to supplement the integral bonding attachment as noted, will prevent disbonds in the honeycomb.

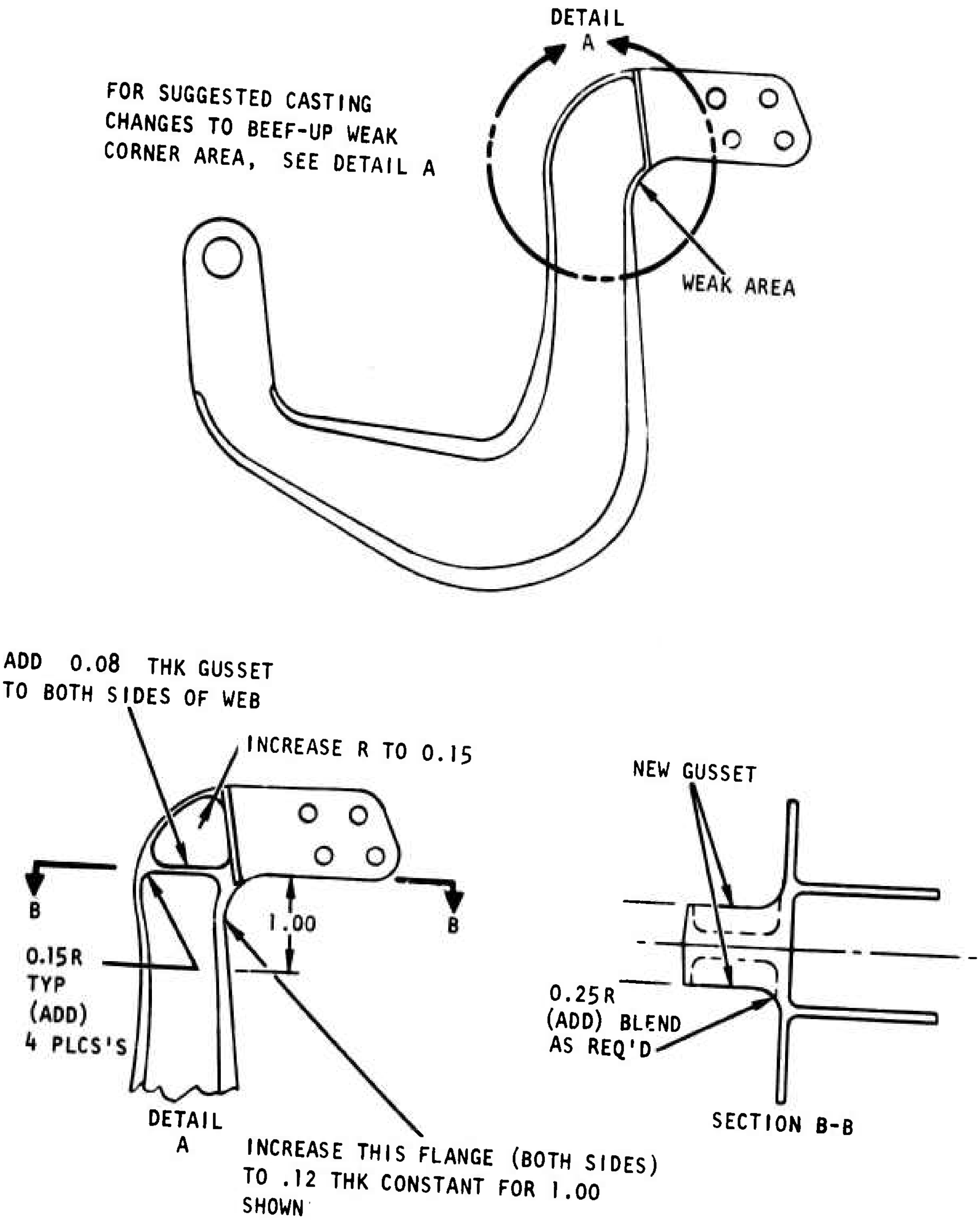


Figure 39. Redesigned Engine Cowl Hinge Repair Concept

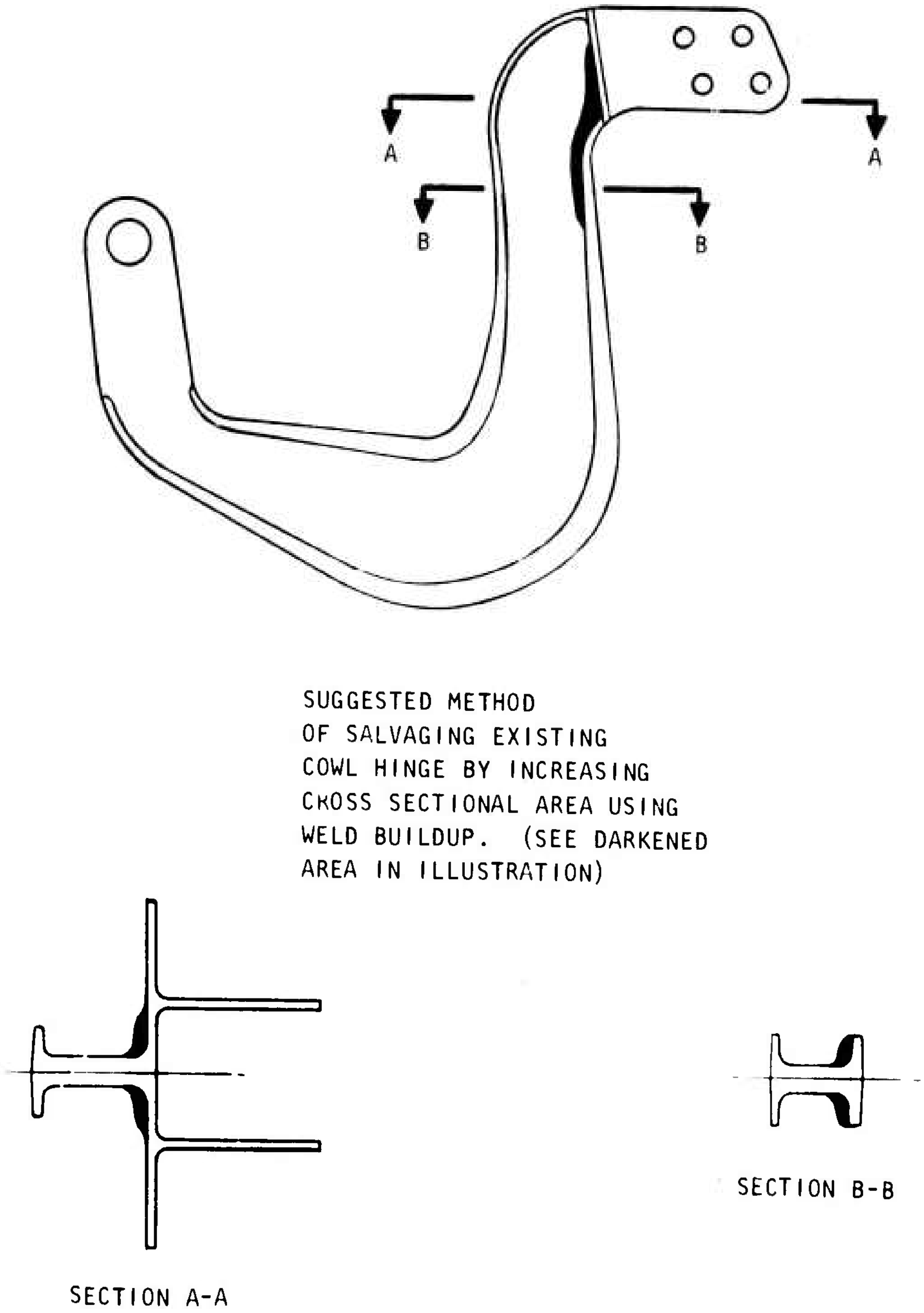
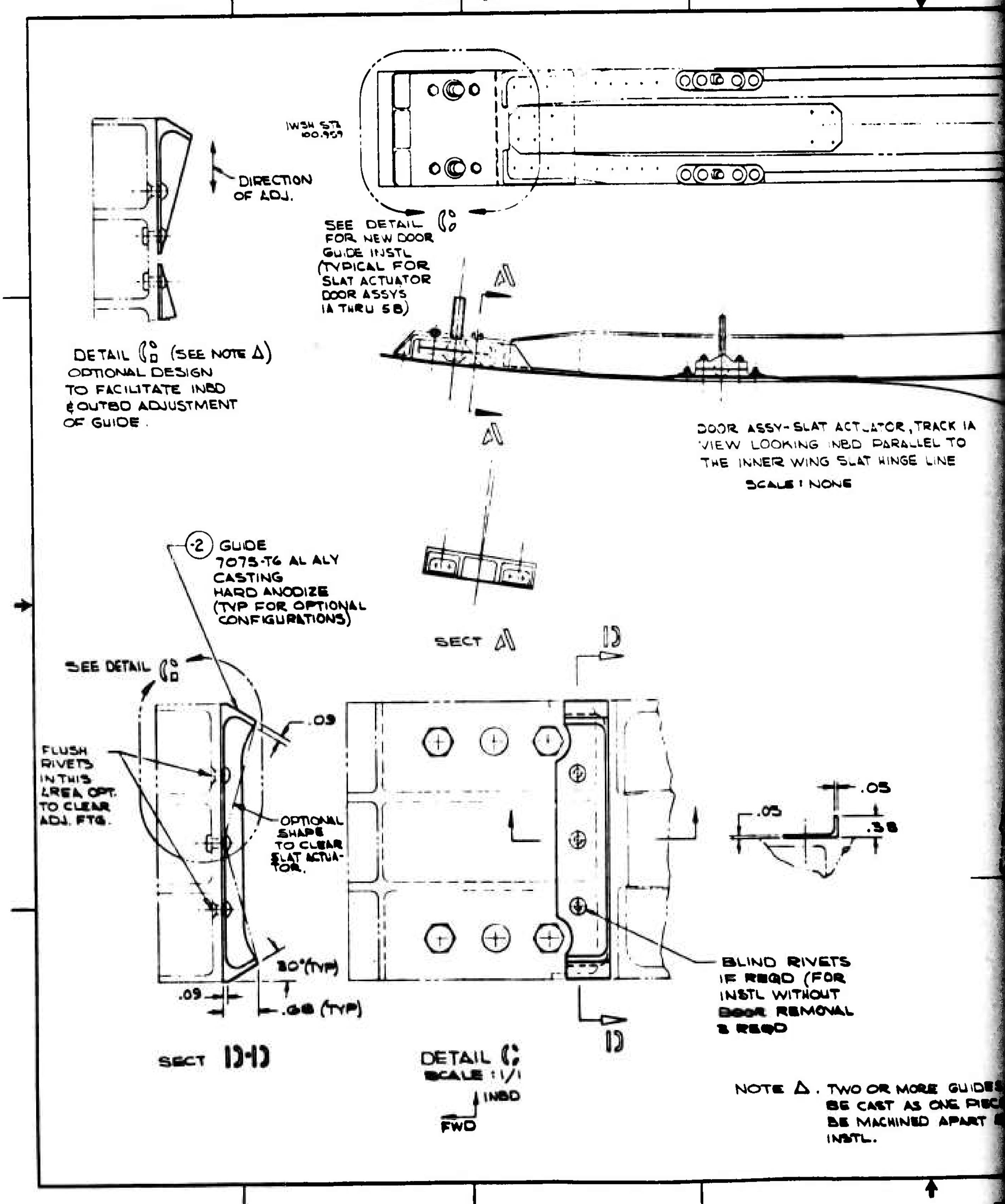


Figure 40. Existing Engine Cowl Hinge Repair Concept


IWSH STA 100.959
DIRECTION OF ADJ.
SEE DETAIL C FOR NEW DOOR GUIDE INSTL (TYPICAL FOR SLAT ACTUATOR DOOR ASSYS IA THRU 5B)
DETAIL C (SEE NOTE Δ) OPTIONAL DESIGN TO FACILITATE INBD & OUTBD ADJUSTMENT OF GUIDE
A
DOOR ASSY-SLAT ACTUATOR, TRACK IA VIEW LOOKING INBD PARALLEL TO THE INNER WING SLAT HINGE LINE
SCALE : NONE
2 GUIDE 7075-T6 AL ALY CASTING HARD ANODIZE (TYP FOR OPTIONAL CONFIGURATIONS)
SECT A
D
SEE DETAIL C
FLUSH RIVETS IN THIS AREA OPT. TO CLEAR ADJ. FTG.
.09
OPTIONAL SHAPE TO CLEAR SLAT ACTUATOR.
30°(TYP)
.09
.68 (TYP)
.05
.05
.38
BLIND RIVETS IF REQD (FOR INSTL WITHOUT DOOR REMOVAL 3 REQD
SECT D-D
DETAIL C
SCALE : 1/1
INBD
FWD
NOTE Δ. TWO OR MORE GUIDES BE CAST AS ONE PIEC BE MACHINED APART INSTL.

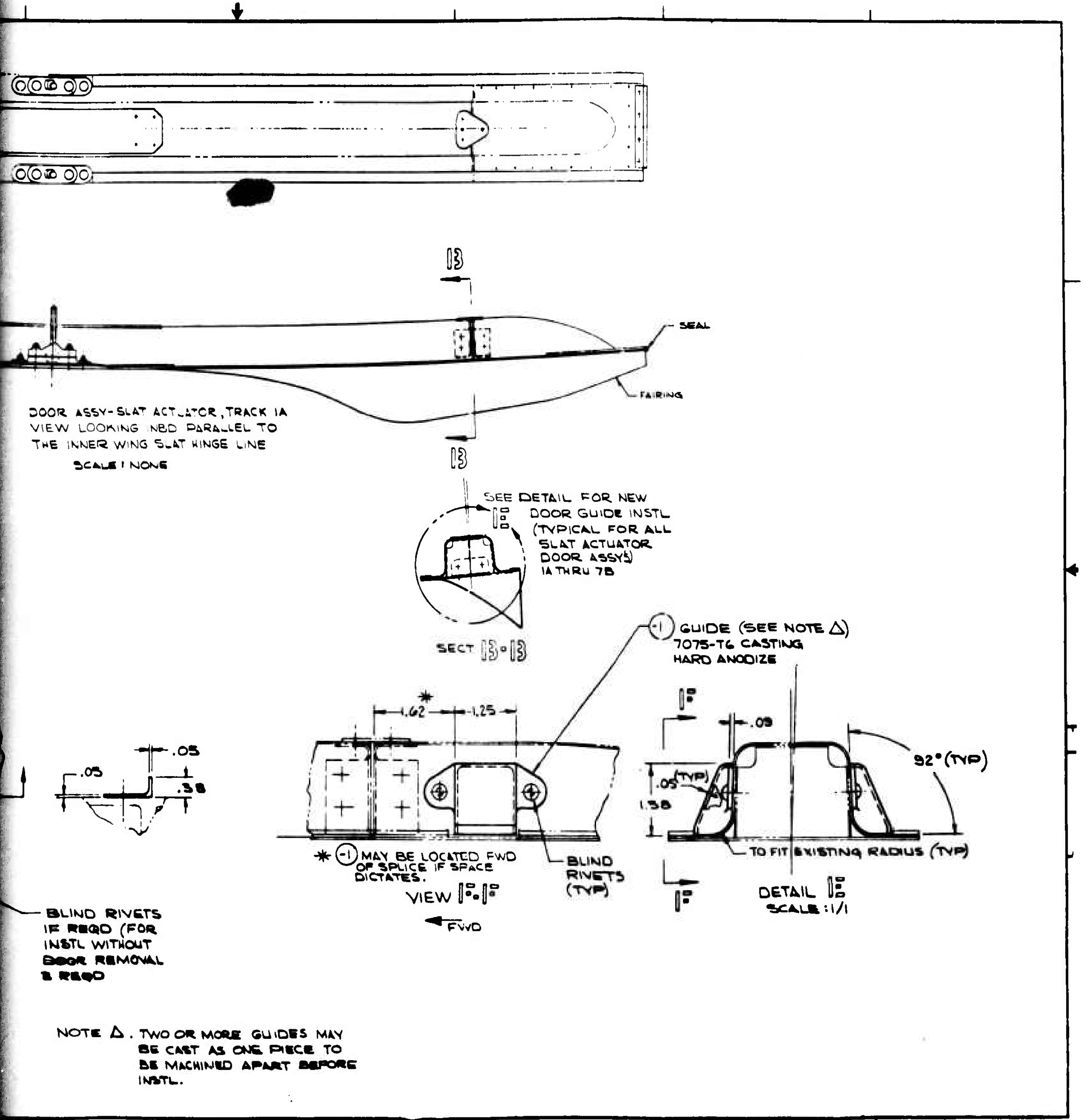
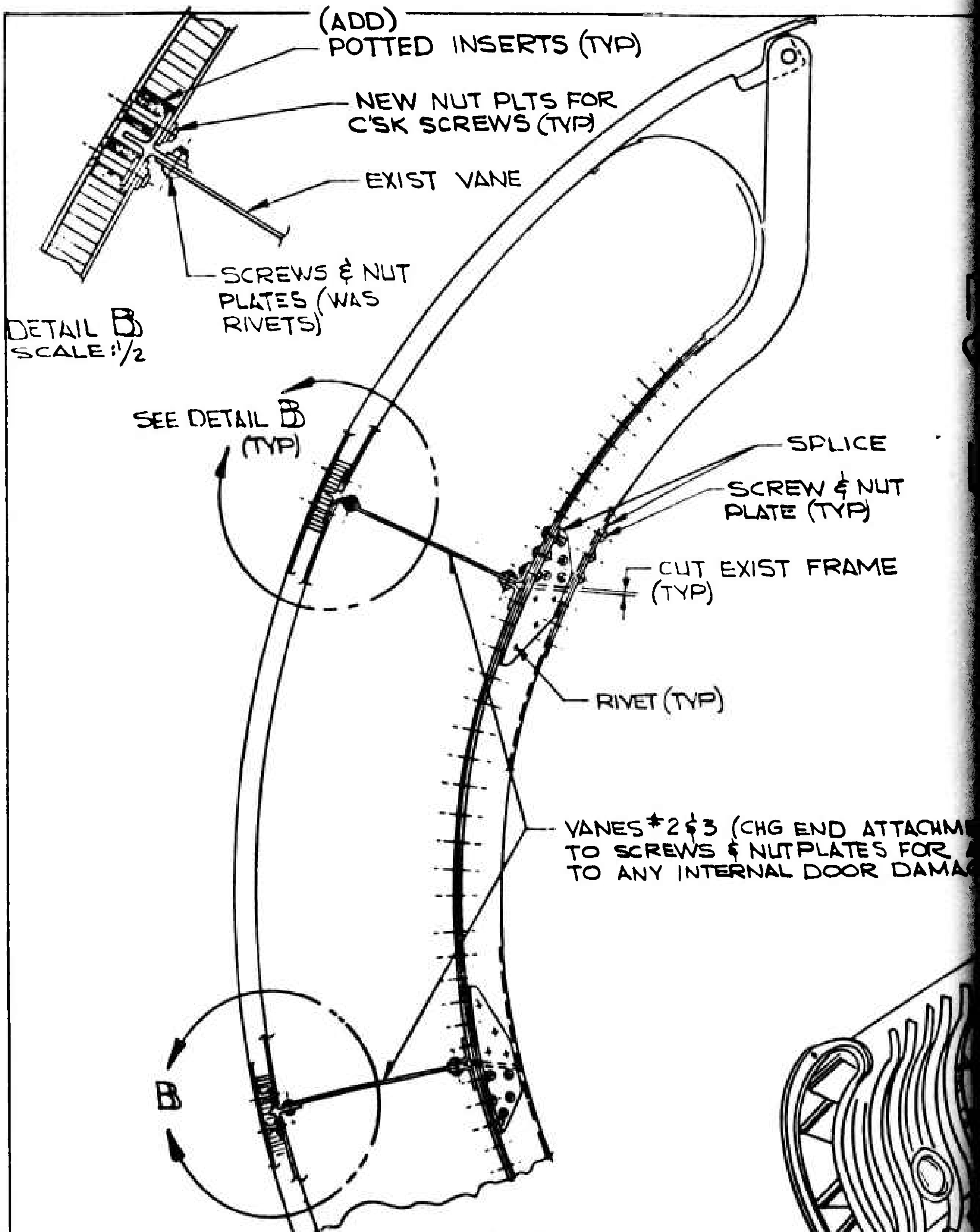


Figure 41. Typical Wing Slat Actuator Door Installation Repair Concept



SECT A-A
SCALE: 1/4

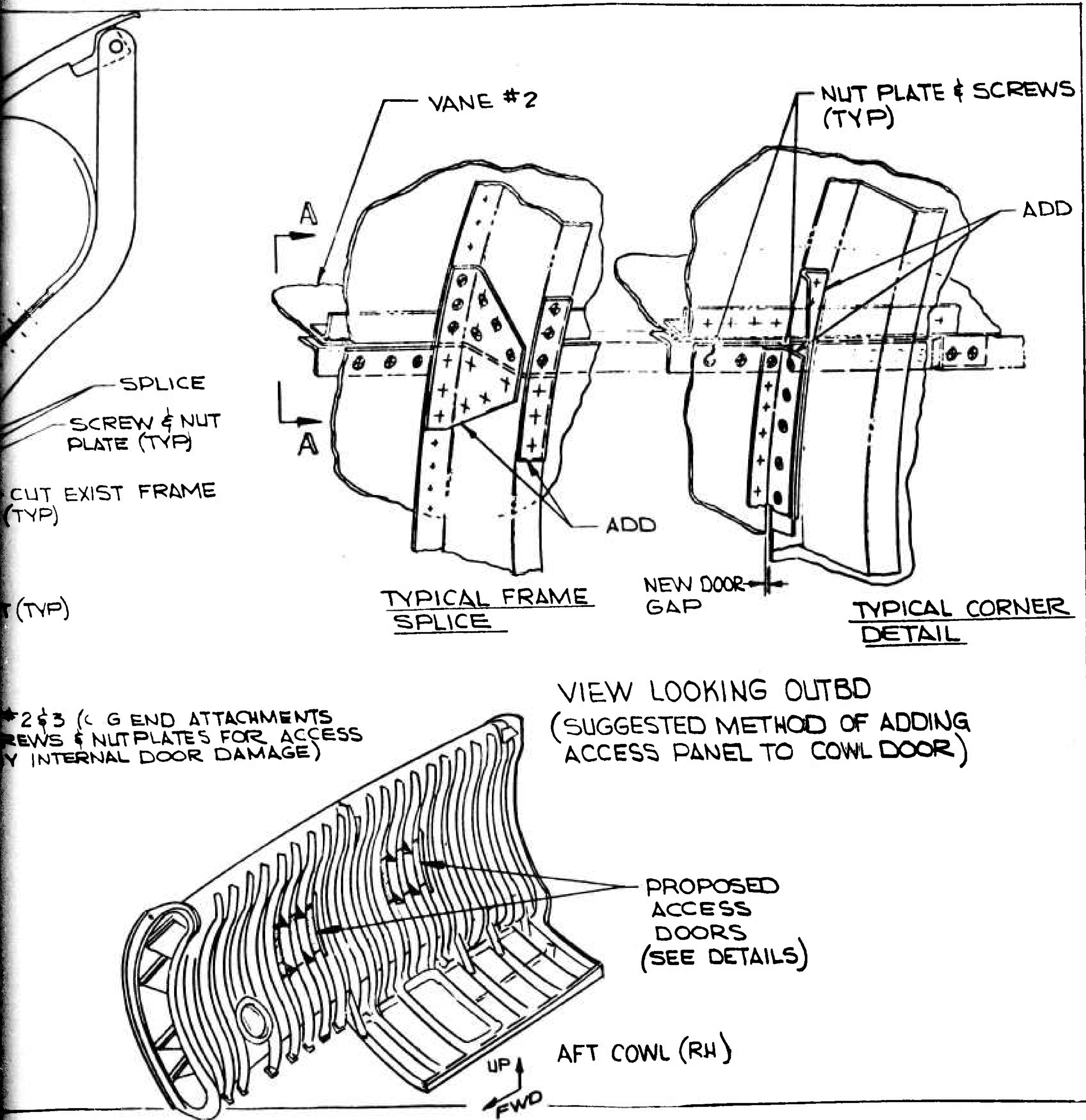
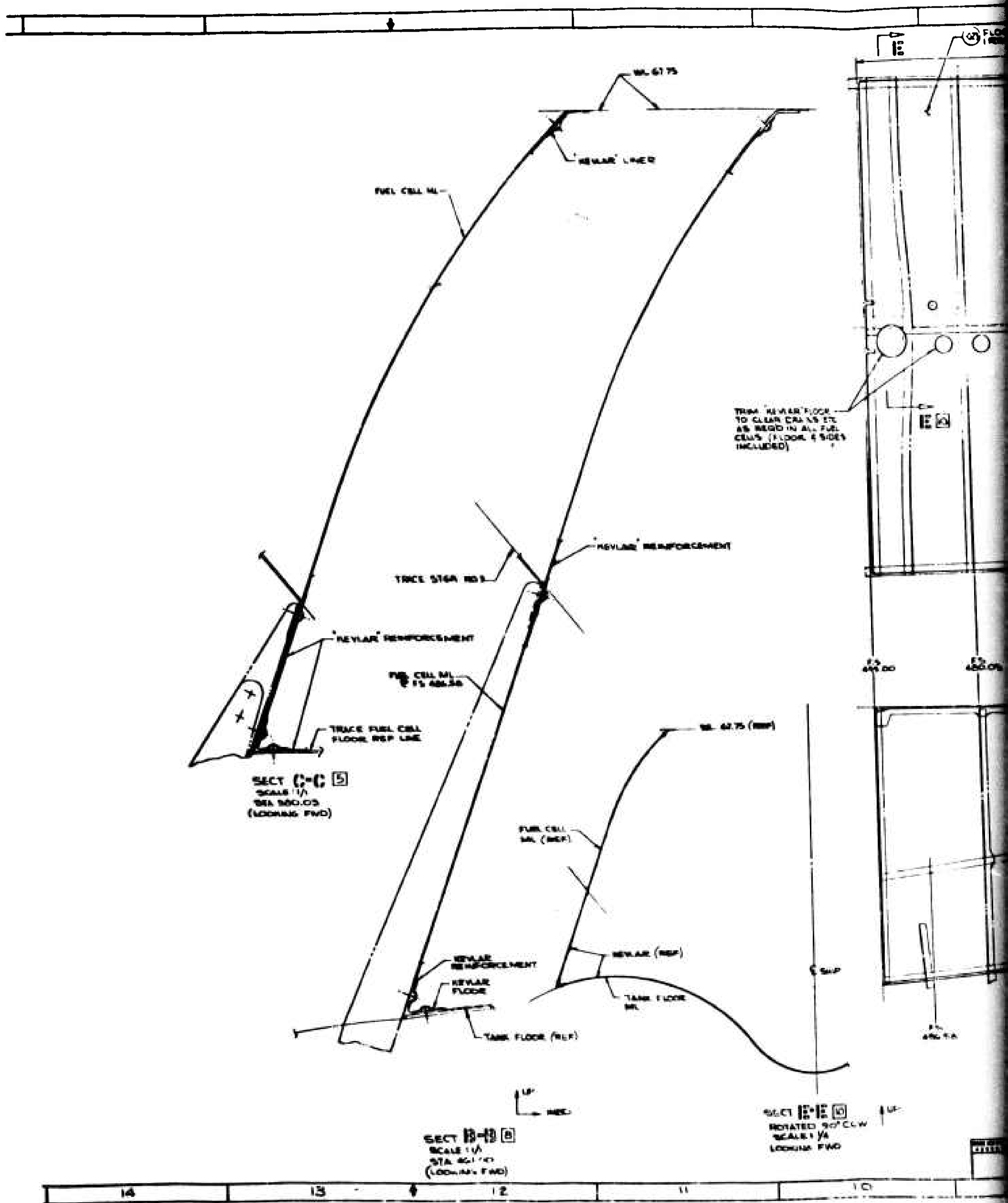


Figure 42. Nacelle Aft Cowl Door Repair Concept



WL 61.75
'KEVLAR' LINER
FUEL CELL ML
'KEVLAR' REINFORCEMENT
TRACE STGR NO.
'KEVLAR' REINFORCEMENT
TRACE FUEL CELL FLOOR REF LINE
SECT C-C 5
SCALE 1/1
(LOOKING FWD)
TRIM 'KEVLAR' FLOOR TO CLEAR CRANKS ETC AS REQD IN ALL FUEL CELLS (FLOOR & SIDES INCLUDED)
FUEL CELL ML (REF)
KEVLAR REINFORCEMENT
KEVLAR FLOOR
TANK FLOOR (REF)
KEVLAR (REF)
TANK FLOOR ML
SECT B-B 8
SCALE 1/1
(LOOKING FWD)
SECT E-E
ROTATED 90° CCW
SCALE 1/4
LOOKING FWD
14
13
12
11
10

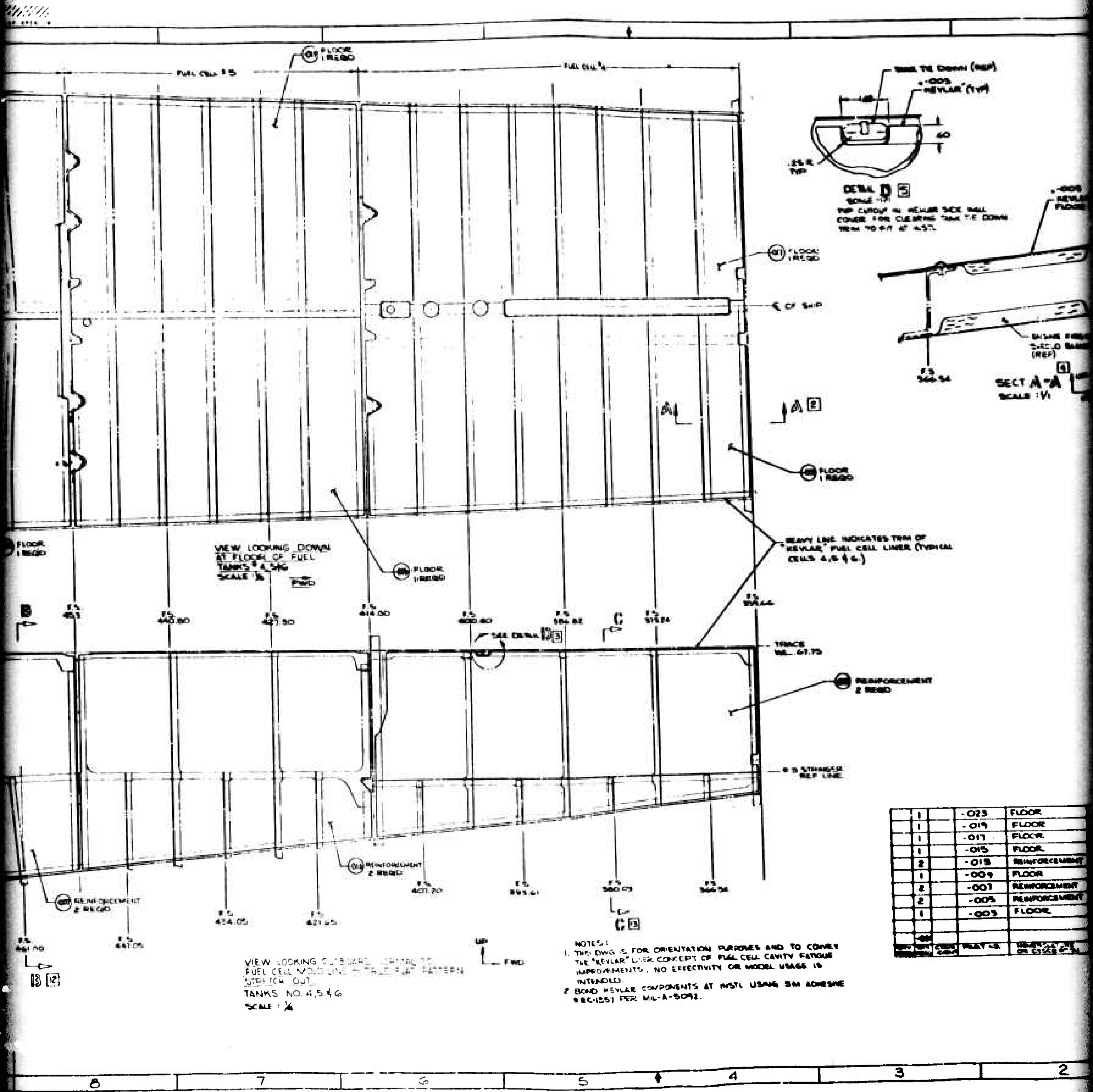


Figure 43. Fuselage Fuel Cell Liner Skins Re

2

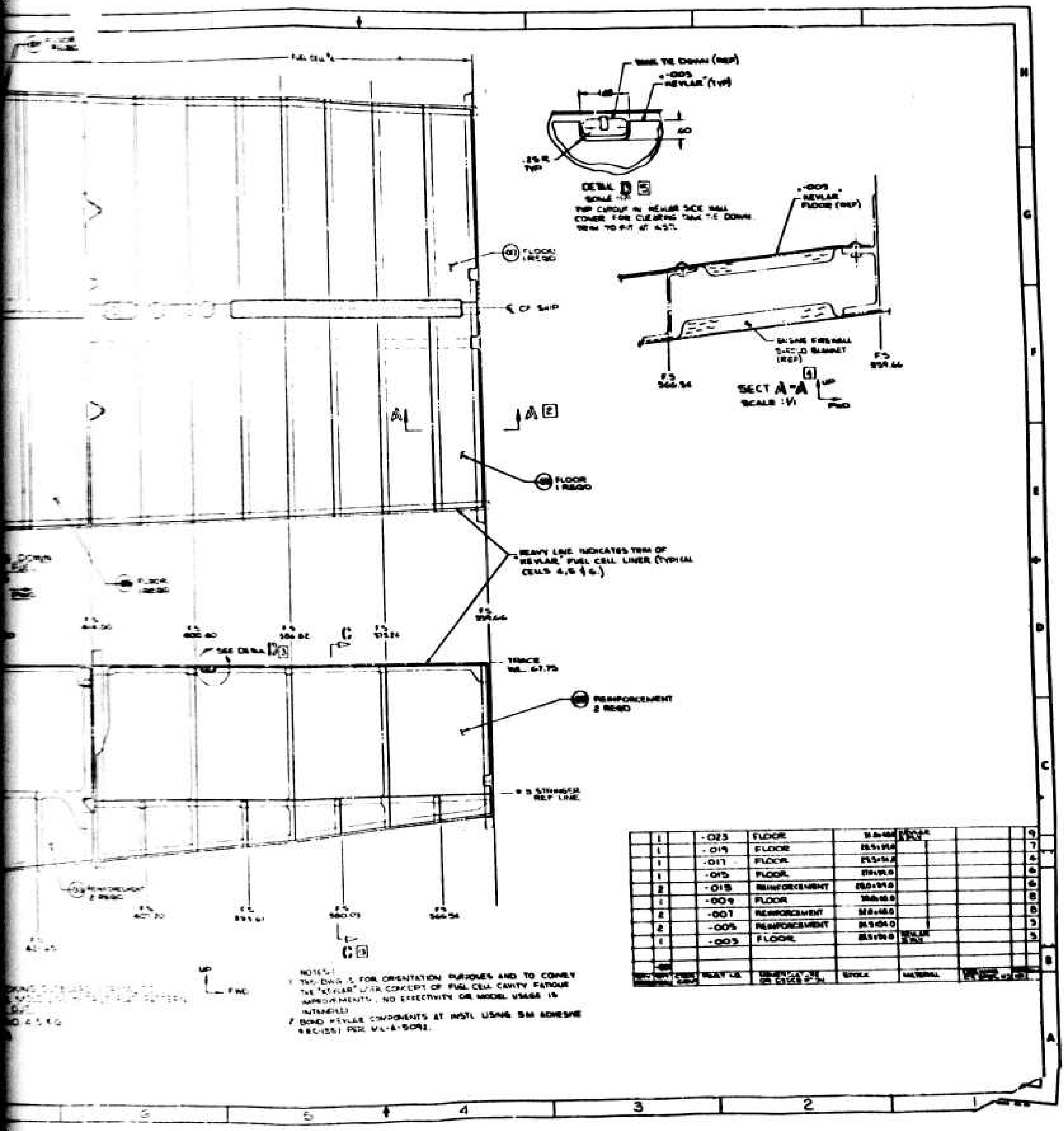


Figure 43. Fuselage Fuel Cell Liner Skins Repair Concept

3

7.3.8 FUSELAGE FUEL CELL LINER SKINS REPAIR CONCEPT (Refer to paragraph 4.8 for problem.)

Kelvar, a woven fabric material produced by DuPont, can be bonded to the fuel cell skins for crack prevention. This approach would involve patching cracks that exist, using standard localized patching methods contained in existing repair manuals, and bonding a Kelvar liner to act as a load-carrying member between frames (see figure 43).

7.4 BATTLE DAMAGE REPAIR

7.4.1 INTRODUCTION

Battle damage repairs are designed for repair of aircraft being operated under combat conditions. Repairs must be simplified to the maximum extent possible, and a wide range of material and fastener substitutions must be provided. All skin repairs should be outside the mold line; that is, they are nonflush to the maximum possible. Material requirements should be to the lowest strength possible, thus providing a wide range of stronger substitute materials. Repair fasteners selected should be those in common use, and generally the lowest-strength fastener (within practical limits), to provide the widest possible range of fastener substitution.

Nonflush repairs should be blended or faired with the skin by chamfering or with aerodynamic filler (see figure 44).

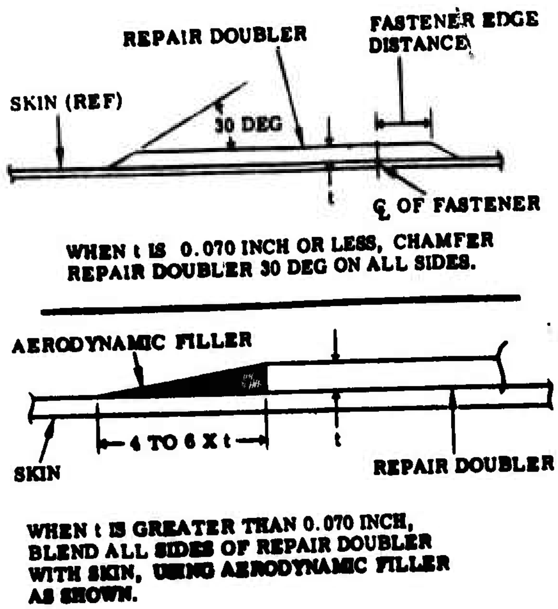


Figure 44. Blending Nonflush Repairs

7.4.2 BATTLE DAMAGE REPAIR EXAMPLES

The following drawings are some examples of battle damage repairs for use in combat areas.

7.4.2.1 Scab Patch - Wet Wing Area

For scab patch for wing skin in wet wing area, see figures 45 and 46.

7.4.2.2 Nonstructural Plug Patch - Wet Wing Area

For nonstructural plug patches for milled skins in a wet wing area, see figure 47.

7.4.2.3 Scab Patch - Between Integral Stringers

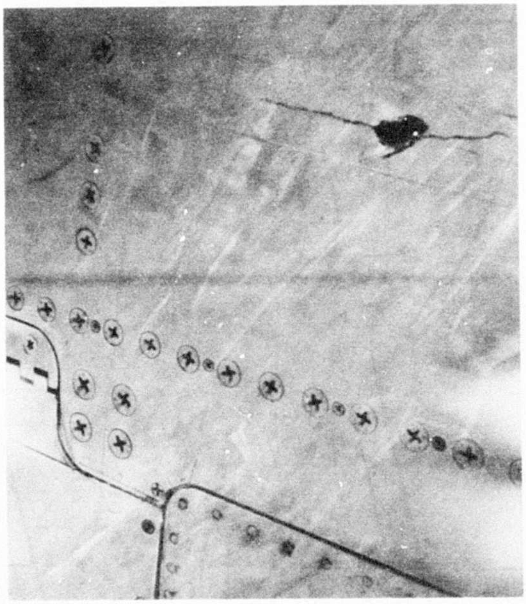
A typical scab patch for skin damage between integral stringers, for horizontal and vertical stabilizer, and wing dry bay area is shown in figure 48.

7.4.2.4 Scab Patch - Across One Integral Stringer

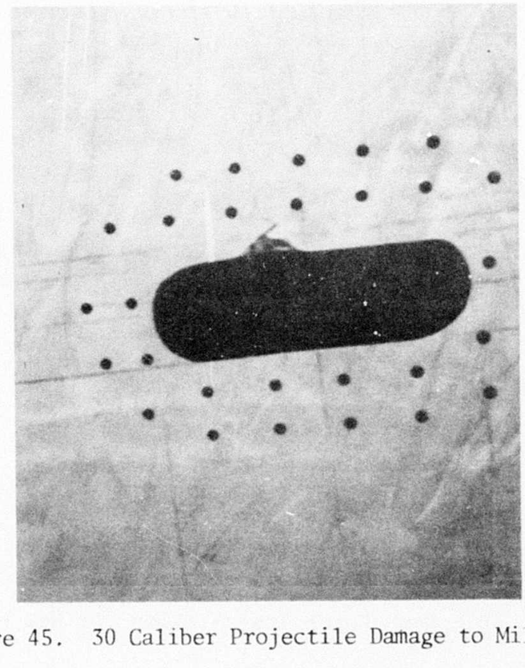
A typical scab patch for skin damage across one integral stringer for horizontal and vertical stabilizer, and wing dry bay area, is shown in figure 49.

7.4.2.5 Scab Patch - Across Two Integral Stringers

A typical scab patch for skin damage across two integral stringers for horizontal and vertical stabilizer, and wing dry bay area, is shown in figure 50.



DAMAGED MILLED SKIN IN WET WING AREA



MILLED SKIN AFTER REMOVAL OF DAMAGED AREA READY FOR SCAB PATCH

Figure 45. 30 Caliber Projectile Damage to Milled Wing Skin

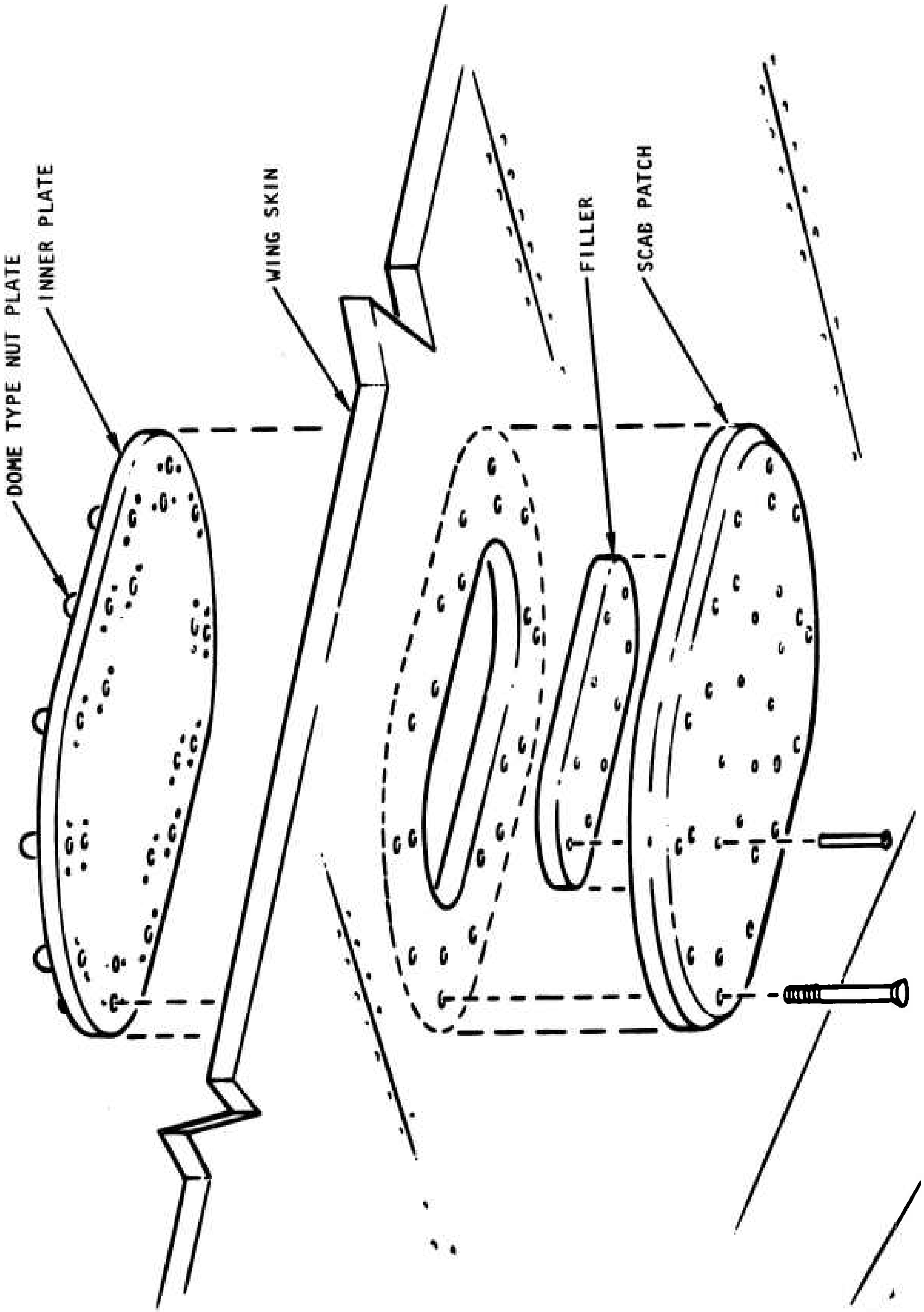


Figure 46. Scab Patch Installation for Wet Wing Area

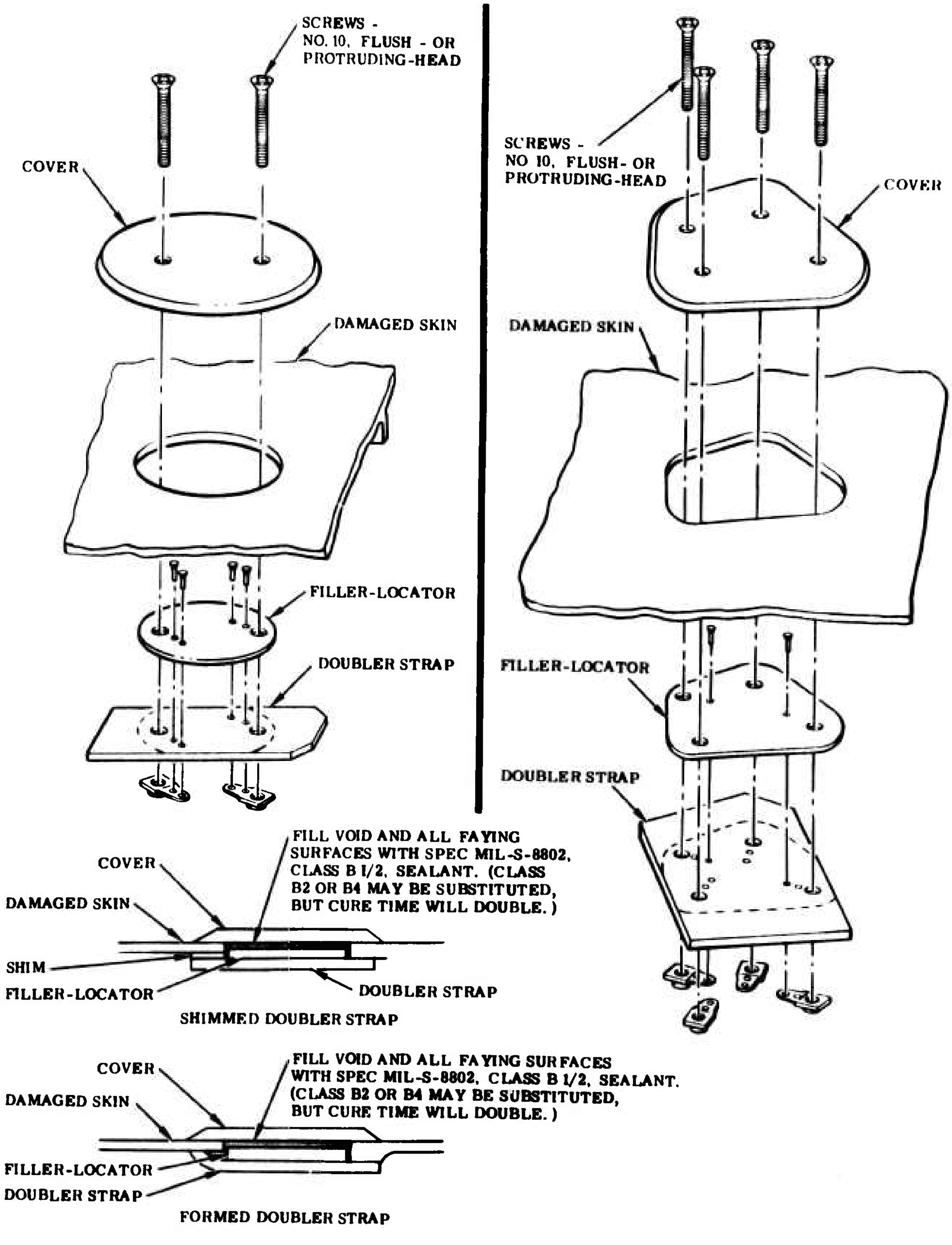
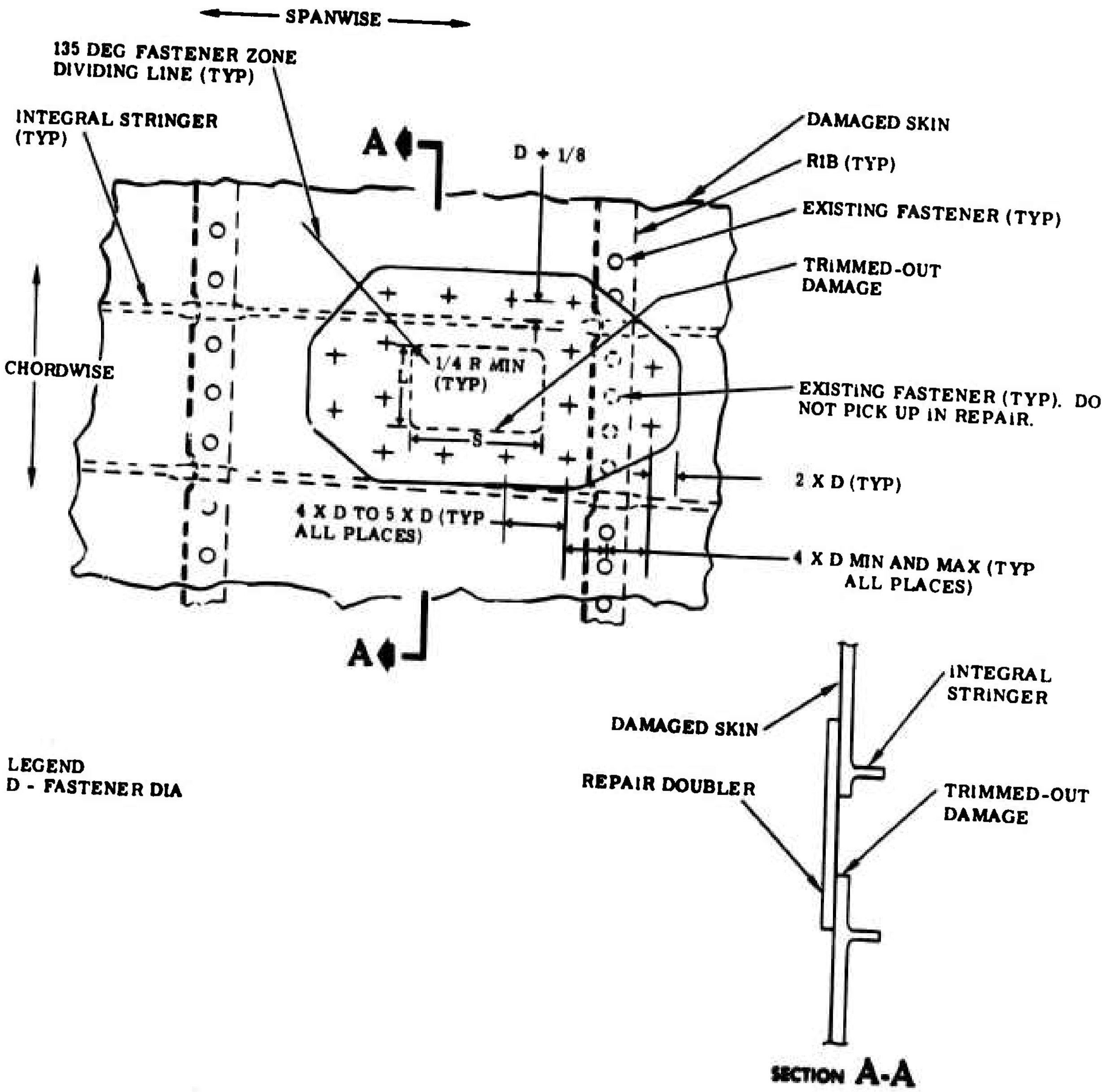


Figure 47. Nonstructural Plug Patches - Wet Wing Area



LEGEND
D - FASTENER DIA

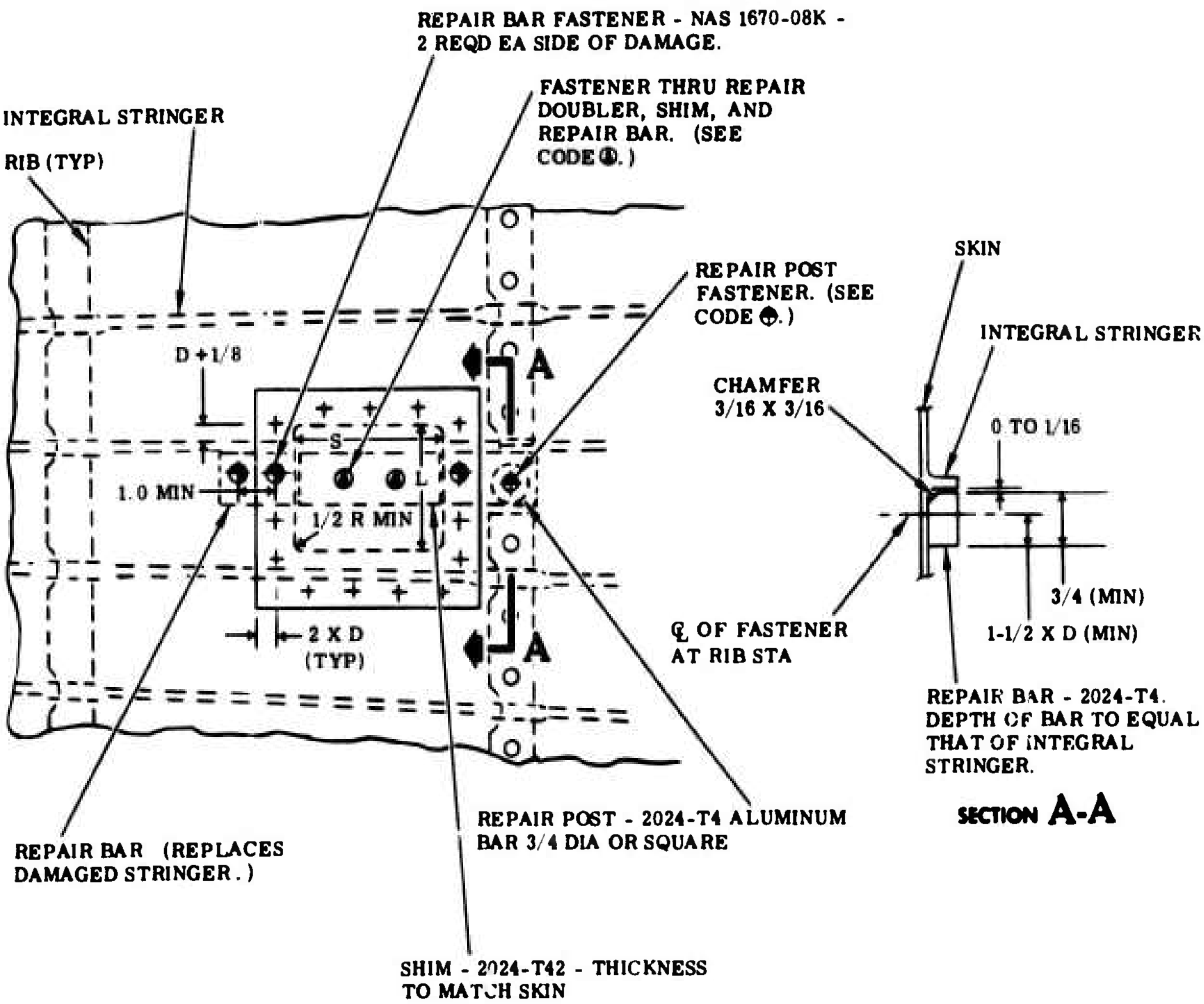
In example 1, a 1-1/2 inch trimmed-out chordwise damage (L) is assumed adjacent to, and outboard of, horizontal stabilizer station 88.16.

1. Repair doubler is 0.375 7075-T6 bare (next standard gage above 0.301).
2. NAS1670-3K Fasteners were selected for repair of chordwise damage (L). The requirement for an "L" of 1-1/2 inches is 4.125 or five fasteners each chordwise side of damage.
3. NAS1670-3K or NAS1670-4K Fasteners could be used as spanwise repair fasteners at the referenced station. NAS1670-3K Fasteners were selected and distributed at 3/4- to 1-inch spacing.

NOTE

When trimmed-out damage is such that the required fasteners cannot be installed between integral stringers and between ribs, extend repair doubler across those members as shown. Do not disturb existing rib fasteners but place repair fasteners as close to rib as possible. See 4 X D MIN AND MAX (TYP ALL PLACES) in sketch. D plus 1/8 inch is minimum fastener spacing from integral stringers because of stringer-to-skin radius.

Figure 48. Scab Patch for Damage Between Integral Stringers



LEGEND

IN EXAMPLE, A REPAIR BAR IS REQUIRED TO REPLACE REMOVED STRINGER. TWO FASTENERS ARE REQUIRED THROUGH SKIN AND REPAIR BAR EACH SIDE OF DAMAGE. WHERE RIB IS TRIMMED OUT TO ACCOMMODATE REPAIR BAR, A REPAIR POST IS REQUIRED. THE FASTENER THROUGH THE REPAIR POST IS COUNTED AS A REPAIR BAR FASTENER.

FASTENER REQUIREMENTS FOR L AND S ARE FOUND AS DESCRIBED IN FIGURE 48. FASTENERS THROUGH REPAIR DOUBLER, SKIN, AND REPAIR BAR ARE COUNTED IN REQUIREMENTS FOR L.

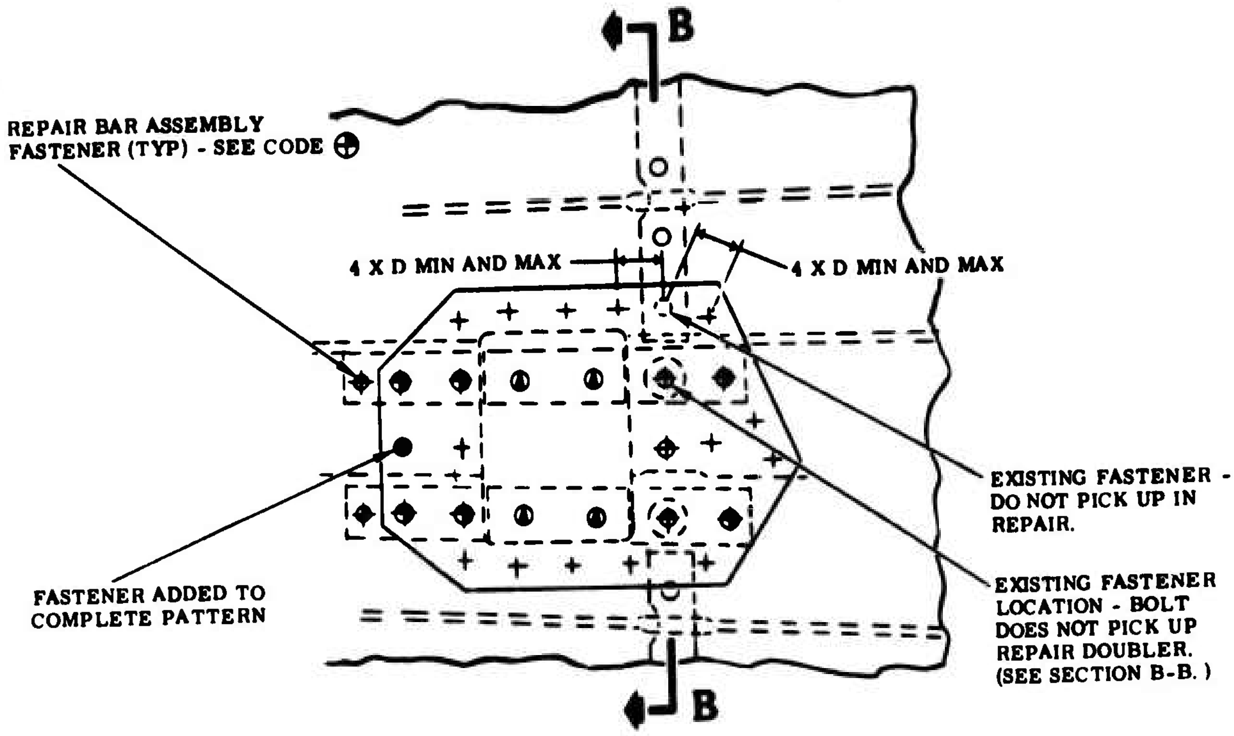
D - FASTENER DIAMETER

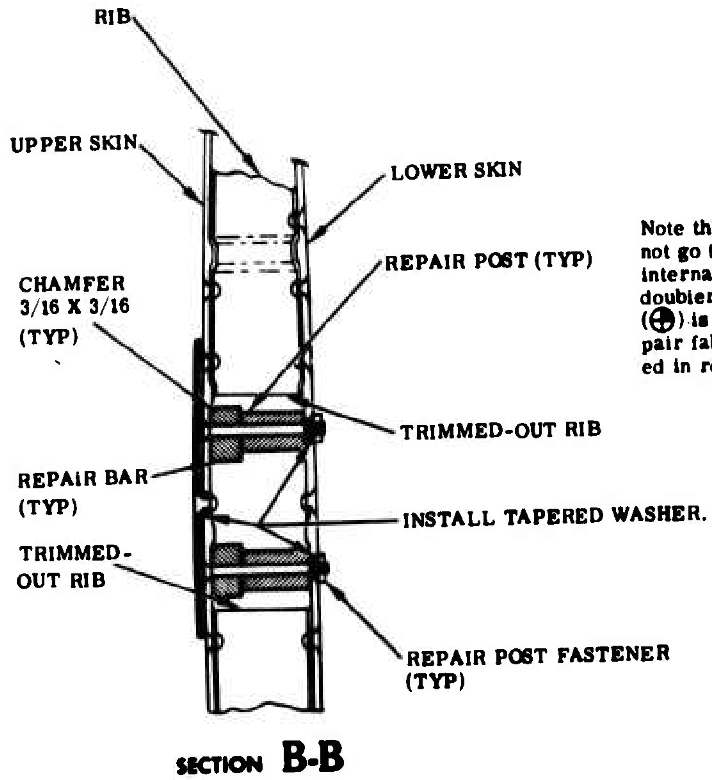
⊕ - AN509 BOLT WITH MS20365 NUT

+ - REPAIR DOUBLER FASTENERS.

Ⓐ - FASTENERS TO MATCH THOSE USED IN SPAN-WISE SIDE (L) OF REPAIR. SPACING IS 4 X D MIN, 8 X D Max.

Figure 49. Scab Patch for Damage Across One Integral Stringer





Note that in this repair example repair post fasteners do not go through repair doubler (to facilitate installation of internal repair members) and are not counted in repair doubler fastener requirements. An assembly fastener (⊕) is required at one end of each repair bar for final repair fabrication and assembly. This fastener is not counted in repair requirements.

LEGEND

D - Fastener diameter

⊕ - MS20426AD3 or any small flush-head rivet

● - AN509 Bolt with MS20365 Nut

⊕ - Fasteners to match those used in spanwise side (L_1) of repair. Spacing is 4 x D min, 8 x D max.

Figure 50. Scab Patch Across Two Integral Stringers

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MIL-STD-1568, Materials and Processes for Corrosion Prevention Control in Aerospace Weapons Systems

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8.7 TECHNICAL ORDERS

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T. O.-1-1-4, Exterior, Insignia and Markings Applicable to USAF Aircraft

T. O. 1F-100D-3DB, F-100D & F Structural Repair Combat Zone

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